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IN-FLIGHT CHECKOUT [U]

PROJECT APOLLO

June 21, 1963

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ABSTRACT

The joint RAND/Bellcomm report is intended to define the basic Apollo CM in-flight checkout requirements, and to specify a manner of testing based on these requirements.

Four major electrical and electronic subsystems in the CM are examined in detail to obtain inferences in testing procedures, number and types of test points, and types of measurements required.

The conclusions drawn from this study are:

1. That the G&N Computer be redesigned using redundancy to achieve greater inherent reliability.

2. A redesign of the PCM telemetry system providing redundancy, by duplication within the same weight limits, may prove to be the preferred way to achieve the desired reliability.

3. For the subsystems examined, and with the computer and PCM systems not included in a maintenance concept:

    a. Confidence testing points, not operationally displayed, be displayed by light indicators driven by individual comparators.

    b. Diagnostic testing be accomplished by stationary, hard-wired, multirange VTVM.

4. The recommended system is achievable if the proposed NAA-IFTS implementation is modified to:

    a. Include stimuli.

    b. Remove comparators from diagnostic test points.

    c. Include more diagnostic test points.

5. If a reliable G&N Computer is available, a better in-flight checkout system may be achieved by supplementing the VTVM system with the PCM Computer configuration to reduce overall crew work load.

IN-FLIGHT CHECKOUT

PROJECT APOLLO

1.0 Introduction

On-board testing will play at least two important roles during the Apollo mission. Test results will be used to make operational decisions, and in the event of equipment failure, on-board testing will be necessary to ascertain the nature of the problem and to assist in any corrective actions. The question that will be addressed is: what should be the nature of the test system that is placed aboard the command module, to assist in the functions of making operational decisions and restoring the spacecraft to a mission-ready condition in the event of equipment failure? These roles are often called confidence testing and diagnostic testing, respectively.

The results presented in this report are based upon a joint Bellcomm/RAND study that was performed during the past 2 1/2 months. The study was done quickly and therefore some of the data is sketchy and some problems that require a thorough analysis were considered only cursorily. In other areas important experimental evidence is lacking and therefore one could only speculate on the outcome of some future programs and developments. For example, it is not really known what the physiological and psychological conditions of the astronauts will be after, say, a week of isolation in space under mission conditions. An attempt will be made to point out the most serious deficiencies of information, to show why an unequivocal solution cannot be presented at this time. However, the underlying reasoning and the nature of the test system that follows from this reasoning will be presented.

1.1 Mission Profile and Roles of Checkout

Let's examine the Apollo mission profile and the roles of testing in that context. In the first Figure (Figure 1), a typical profile for part of a lunar landing mission is depicted. The significant events in the mission are indicated, as are the anticipated time-loading of the three astronauts--for mission required tasks other than maintenance and checkout, per se.

A decision must be made prior to each of the significant mission events. The question of whether to proceed with the mission or abort must be answered, and if the mission is to proceed, the astronauts must decide on how. That is, they must select a mode to employ and the on-board equipments to use during at least the next mission phase. Decisions such as these must also be made between the indicated points--in fact, they must be made continuously.

1.1.1 Confidence Testing

Let's examine one of these decision points paying particular attention to the nature of the decision and the role of checkout in making this decision; this role is called confidence testing or status evaluation. Let's assume that the mission has progressed to the point where the crew is preparing to enter lunar orbit. At this point the crew has information concerning the events that brought them to this point. They know about past repairs and corrections, and they may know about past errors and failures. They have information about the quantity of fuel and life support remaining and they should know what this information means in terms of mission capabilities. At this point they must look ahead and select a path to follow. This means that they must select a set of spacecraft equipments and ground operations that they will place primary reliance upon.

Schematically, the selection of a continuation path looks like this (Figure 2). At this point in the mission, these are the equipments that could be used for the next two mission phases. Their relationships with respect to primary or secondary nature, or series of parallel arrangements are also shown. This Figure will not be discussed in detail, but the point is that they must select a path of equipment to rely upon and this selection must be made within the existing operational context. This is the decision problem, and it clearly is not an easy one. It deserves and will be given further analytical treatment.

For the purpose of this study, an answer has been assumed to this problem. The assumption is that each decision will be made using the simple operational decision rule of: search and find an acceptable open path for the next mission phase. This means that the crew must obtain data on each of the equipments which could be used, determine which of them could be relied upon to perform during the next mission phase, and select a workable set from these. Assuming that the crew is explicitly trained for this decision-making task, and this appears quite reasonable, then there is no reason to expect, at the present time, that they could not function adequately in this role. The confidence testing role of the on-board checkout system is to provide the necessary information on the equipments' conditions--both current and projected. Flight displays provide much of this information, simulated operations using flight controls and displays would provide much of the remaining information, and later in this report consideration will be given to ways of augmenting the flight displays to provide an adequate confidence testing capability.

This points out a ground rule that has been adopted. Ground rule: when planning the checkout system, maximum use should be made of data that is available in the spacecraft for other purposes. This data is essentially free for the purpose of checkout.

1.1.2 Diagnostic Testing

Let's go back to the mission profile (Figure 1) and consider the other role of checkout equipment -- that of assisting in the maintenance function. Before considering diagnostic testing within the maintenance function, however, let's first examine the potential need for maintenance, the anticipated astronauts' capabilities for testing, diagnosis, and repair, and the time during which maintenance could be performed. Each of these contextual issues will be discussed; other sections will discuss particular vehicle equipments and their maintenance and testing needs.

The need for maintenance arises out of equipment failures, and the number of failures that can be anticipated depends very much upon whose reliability estimates one uses. Using present contractors' estimates for example, it is estimated that the probability of at least one critical failure during the mission is on the order of 50%, and the probability of 2 or 3 failures is on the order of 10%. At this point in the program, however, one can select reliability estimates to support almost any position. The objective is not to play the numbers game, but merely to explicate the almost obvious point that, using the present design concepts, in-flight maintenance will be required for a successful mission, and from this to show the need for test equipment and spares.

In fact, this study has found that, using present design concepts, all of the electronic systems in the command module will require maintenance and spares to meet their reliability goals. Some of these failures could be avoided by using more built-in redundancy; in at least one case, less redundancy appears desirable, and an examination of these provisions in detail is contained in subsequent sections. The balance will require either functional redundancy, so that the mission could continue in spite of the failure, or spare parts and a maintenance capability.

It is also clear that the probability of mission success will depend on the type and quantity of spares available. They will not all be used, but they must be available when needed. The more spares--the higher the probability of success. Therefore, since weight allocated to test equipment will probably detract from the weight allowable for spares, one should search for the lightest test system, that in conjunction with the spares, will satisfy the maintenance requirements.

Considering the crew's capability to perform maintenance, it seems reasonable for the Office of Manned Space Flight to insist that at least one crew member should be technically competent to perform fault diagnosis and module replacement--when assisted by the test system. In keeping with the self-sufficiency concept, it is not unreasonable to require the other crew members, as well, to be trained for maintenance operations. Provided that this training in maintenance operations is given, and provided that work-rest cycles are observed so that the crew is not excessively fatigued, then it appears that the astronauts would be physically capable of performing mental, interpretive maintenance operations such as those inherent to fault diagnosis, except possibly in periods of extreme time pressure when they would be forced to rely upon back-up systems and alternate modes. If provision is made for physical stability during maintenance, that is, if body supports and attachments are provided, then a growing body of experimental evidence indicates (but not conclusively, as yet) that the astronauts should be physically capable of performing maintenance operations such as to remove and replace, even small modules, and to manipulate such items as bits and pieces of electrical and electronic equipment. The task of the test equipment is to assist in failure recognition and isolation, that is, it must be effective for diagnostic testing. There are several kinds of test systems that could do this and they will be described and evaluated later.

Allusion has been made to the times within which maintenance could occur, but have not been covered explicitly. It can be seen what these times are on this typical mission profile (Figure 1) which shows time patterns of crew rest and work. Note that the time-critical points, during which the astronauts would be under the greatest psychological stresses, are in earth orbit, where they are fresh, can rely upon earth for assistance and could take a second orbit if necessary, and among other points, immediately prior to each phase initiation. With respect to the time for maintenance, then, one must plan the test system to assist in finding a faulty part within the allowable time and to the extent that time constraints and psychological stress would preclude a reliable maintenance operation, provision for redundant systems or alternate modes must be made.

What are these allowable times? Data has been extracted from these time lines into representative times available for checkout and repair prior to important mission events (Figure 3). The times listed under "in couches" are the durations during which the astronauts would be constrained in their couches, but would have time to perform checkout functions. The times listed as "at bay" are the times during which the crew could be working freely at the instrument bays within the command module. It appears that the time available for checkout only would be at least 20 minutes in each instance, and the more critical time--that for maintenance--appears to be of the order of one hour or more. Therefore, to be considered, a test system must provide the capability to make status evaluations and phase initiation decisions in at most 15 to 20 minutes and repair of a failed system in about an hour. Each of the alternative systems that will be presented later will provide these capabilities.

1.2 LEM Interaction and Roles of Earth

Before initiating a system-by-system discussion of test and maintenance needs and possibilities, there are two more background and contextual topics that will be discussed briefly. First, there is the matter of the interaction between the LEM and the CM/SM checkout system. During the study the LEM has been generally neglected, partly because checkout of the LEM using CM equipment is not part of the current Apollo plan. Realize, however, that further study might show it to be desirable to check on the status of some LEM equipments using the CM checkout system. In that event, since the equipments to be placed aboard the LEM should be of the same general nature as those of the CM/SM, it appears that any test system that could handle CM/SM equipments could, with high likelihood, be augmented and made capable of some remote LEM checkout.

A few of the desirable checkout and maintenance roles of earth will be considered in the following sections. In general, however, the test systems described are consistent with the concept of crew self-sufficiency. Nevertheless, the Earth stations could play a significant support role for checkout and maintenance. Given that much of the spacecraft data obtained during normal operations will be telemetered to earth, and considering that the data bank and pre-launch checkout data will be available at IMCC, there are several support roles that are obviously desirable. One of these is the constant evaluation of data to ascertain the existence of harmful trends. Another is the use of systems experts to assist the astronauts in locating and alleviating the causes of equipment problems.

So much for background and context; consideration will now be given to the test and maintenance needs of four groups of CM/SM equipment that collectively constitute 80 to 90 per cent of the electronics and most of the electrical equipment. Then several alternative test systems will be presented and their relative merits, shortcomings, and costs discussed to show which of the systems is preferred.

2.0 System Discussions

This section is concerned with in-flight testing of the Stabilization and Control System (SCS), the Electrical Power System, the Communications and Data Subsystem and the Guidance and Navigation System in the Command Module (CM). An attempt is made to outline reasonable in-flight test concepts, specifically for confidence and diagnostic testing, and to indicate the implications for hardware implementation in accordance with the concepts. The specific recommended implementation is discussed in Section 3.0.

Particular note should be made of the fact that the systems are in relatively early development stages, and are undergoing significant changes as designs progress. Thus, the data and designs referenced in this report must be recognized as preliminary and subject to change, and should not be used out of this context. The arguments presented are accordingly substantially qualitative in nature.

2.1 Stabilization and Control System

2.1.1 System Description

The SCS in the CM is instrumental in the performance of rate damping during launch abort, three axis attitude control, three axis translation, propulsion, and display of spacecraft dynamics. SCS control is activated for flight phases involving the combined CM-Service Module (SM) and the CM alone.

Attitude control and translational control are accomplished through SCS outputs exciting the Reaction Control System (RCS) jet relay coils. References for attitude maneuvering are derived from the Inertial Platform (IMU) of the Guidance and Navigation System (G&N), or from the body-mounted attitude gyros (BMAG's) within the SCS.

Firing time and thrust vector direction for the Service Propulsion System (SPS) are established by SCS control outputs. Specifically, pitch and yaw reference signals for engine gimbal positioning are processed in the SCS in thrust vector control (TVC) circuits. Firing-duration signals from the G&N system (via SCS) are backed up in the SCS by an x-axis accelerometer, processing electronics, and displays of velocities remaining. Figure 4 illustrates the functional content of a single channel.

Three major operational displays are provided as part of the SCS. The Flight Director's Attitude Indicator (FDAI) provides continuous display of spacecraft attitude, attitude error, and body rates in three axes. The Gimbal Position Indicator (GPI) displays SM engine gimbal position in pitch and yaw. The ΔV indicator displays the remaining velocity increment to be achieved during a propulsion phase.

The SCS is packaged in 12 subsystems as indicated by Figure 5. Major portions of the electronics are contained in the five Electronic Control Assemblies (ECA's). Modular construction is utilized throughout, with all 12 subsystems being designed for either module or total replacement, given access within the spacecraft. Currently, only the FDAI is inaccessible for replacement, although design changes in the astronaut's control panel are being studied which may alter the situation. Figure 5 also indicates the subsystem weights, volumes, modular content, and potential duty cycles.

Minneapolis-Honeywell (M-H) studies currently indicate an approximate average module replacement time of 40 minutes for the ECA's. Rate and attitude gyro replacement times are in the order of 5 to 20 minutes, although 20 minutes warm-up is required for the attitude gyros.

M-H currently is estimating the overall system MTBF at 650 hours. Based on 150 hours operation, reliability is in the order of 0.8. The electromechanical devices, particularly the FDAI and attitude gyros appear to be pacing the overall reliability.

2.1.2 Mission Profile

For purposes of SCS operation, the following mission increments appear logically suited to test oriented discussions:

1. Launch

2. Earth Orbit to S-IVB Jettison

3. Translunar Flight

    a. Orientations, Alignments, Sightings

    b. Mid-course Corrections

    c. "Routine" Attitude Maneuvering

4. Lunar Injection

5. Lunar Orbit
    a. Orientations, Alignments, Sightings
    b. Lunar Observation
    c. LEM Rendezvous and Docking
    d. "Routine" Attitude Maneuvering
6. Transearth Injection
7. Transearth Flight
    a. Orientations, Alignments, Sightings
    b. Mid-course Corrections
    c. "Routine" Attitude Maneuvering
8. Re-entry

The probable mode of SCS operation during each of the listed increments is discussed in the following sections. This does not represent a detailed study of SCS profiles, but is considered sufficient for later identification of types of events significant to testing philosophies.

1. Launch - SCS primary mode of operation will be in the monitor mode during launch. The FDAI will display total spacecraft attitude and angular rates. Attitude error is not displayed due to differences in G&N and S-IVB guidance equations. No SCS control functions are required during a nominal launch.

    Astronauts will be restrained during the launch phase, thus preventing any activities other than those immediately available at the control panels.

2. Earth Orbit to S-IVB Jettison - Current design implies continued SCS operation in the monitor mode during earth orbit. Interlocks inhibit RCS jet firing until the S-IVB has been jettisoned. Such a restraint is of particular note for testing considerations, since any form of controlled dynamic testing is ruled out.

However, since sighting and alignment requirements exist during earth orbit, some form of manual control of the CM/SM/S-IVB will be required. Use of the SCS control electronics, at least partially, for firing S-IVB reaction jets has been suggested recently.* Such an implementation could affect test philosophies, from the standpoint of equipment availability for testing, and types of tests to be considered.

3. Translunar

   a. Orientations, Alignments, Sightings - Primary SCS mode for these functions is G&N attitude hold. The number of such operations is not well defined, but they should occupy an appreciable portion of the mission. In general, the SCS attitude control mode may also be used to perform similar maneuvers.

   b. Mid-course Corrections - Mid-course corrections will generally be preceded by a series of sightings and alignments in the G&N or SCS attitude control mode. The G&N ΔV mode will be the primary propulsion mode. Total operating time in these phases will be relatively short, in the order of seconds.

   c. "Routine" Attitude Maneuvering - During "routine" attitude maneuvers; that is, when G&N is not available, or sighting and alignment requirements are less severe, the primary operating mode is expected to be SCS attitude control.

4. Lunar Injection - Primary operating mode is G&N ΔV.

5. Lunar Orbit - G&N and SCS attitude control will be required periodically for sightings, alignments, and orientations, in the manner previously noted. The SCS local vertical mode will be required periodically for visual observations of the lunar surface. Manual x, y, z translations in either the G&N or SCS attitude control mode will be required during docking maneuvers.

6. Transearth Injection - Similar to lunar injection.

7. Transearth Flight - Similar to translunar flight.

8. Re-entry - Prime mode is G&N re-entry.

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*Mechanization of the Apollo S-IVB/Spacecraft Guidance and Control Interface, dated 21 May 1963, George C. Marshall Space Flight Center, Huntsville, Alabama.

It is noted that at least one alternate mode exists within the SCS for all control modes. The attitude control alternate modes may present some degradation in performance, but are at least sufficient in all cases to permit continued flight until a repair can be made. The alternate ΔV and re-entry modes also may represent a degraded performance level.

In addition to the alternate mode capability, limited parallel redundancy exists. Specifically, thrust vector control circuits employ parallel power amplifiers and control amplifiers, and BMAG electronics and switching capabilities exist to permit rate gyro backup via the attitude gyros.

2.1.3 Confidence Testing Events

From observation of nominal mission profiles, four classes of "events" are identifiable during or prior to which confidence testing should be considered. These "events" are as follows:

1. Earth Orbit

2. Attitude Maneuvers

3. Major Propulsion Events

4. Re-entry

During earth orbit, it is desirable to subject the SCS to a very thorough exercise of all its functions. This requirement is dictated primarily by the consideration of the possible effects of the stress periods of launch and ascent. Any decision for further commitment obviously requires evaluation of total spacecraft status subsequent to the stress period. Such an evaluation implies a checkout sequence.

Furthermore, since the SCS may be required for attitude control in abort situations during translunar injection, entering this mission phase without verification of SCS integrity could present the astronaut with additional decisions as to alternates in an already "panic" situation.

Also, the purely emotional factors involving "peace of mind" of the astronaut and ground personnel can be best satisfied by thorough earth orbit checkout prior to deep-space commitment.

Particular emphasis has been placed on earth orbit checkout because the current SCS configuration is not "flyable" during earth orbit due to fuel constraints and to inhibit-circuits preventing RCS activation until S-IVB separation. However, as noted in Paragraph 2.1.2, this configuration is likely to change. Maneuvering the vehicle for test purposes only, however, will involve fuel penalties. Thus, it is logical to expect that some form of static open-loop earth orbit confidence checking is required other than implicit confidence checking through operational maneuvers.

After leaving earth orbit and separation of S-IVB, attitude control in some form will be required almost continuously throughout the mission. Although task loads and mission profiles are not determined in detail, it is difficult to establish a sequence of probable events in which an attitude control failure during the repeated series of maneuvers for sightings, orientations, alignments, etc. will not permit either continuation of the maneuver in an alternate mode, or postponement of the maneuver until a repair is effected. For this reason, a formal confidence test program for attitude control circuits is not considered a necessary inclusion in the workload after leaving earth orbit. The implicit "flying" status information obtained without expenditure of additional fuel or electrical power for test appears adequate. A possible exception to this approach is the desire to assure that the RCS jets are not continuously firing in opposition to each other due to a malfunction. It is doubtful that the astronauts would be aware of such a malfunction, without supplementary display.

Prior to attempting a major propulsion event, it is anticipated that a form of "countdown" will be accomplished, wherein the spacecraft attitude is adjusted, either by SCS or G&N mode, and the engine gimbal is similarly positioned. The astronaut should then be able to verify the "reasonableness" of the positions assumed prior to initiating the firing sequence. Thus, implicit confidence testing will have been performed operationally except for actual firing circuits. The adequacy of this approach is dependent upon time criticality and possible alternate courses of action should a failure be detected during the "countdown". Time is critical since delays in ΔV initiation will likely result in fuel penalties. However, redundant circuits and alternate modes dilute the repair time criticality. Nevertheless, the checkout capability provided for earth orbit testing could be utilized at least to verify the back-ups in addition to the implicit primary mode checkout.

M-H has noted that failures of engine gimbal position circuits resulting in erroneous gimbal angles during long ΔV's (i.e., lunar insertion, transearth injection) could result in catastrophic tumbling, if not detected, and engine shutdown effected expediently. Consequently, additional operational displays for continuous monitor of critical error signals are under study by M-H. It appears that such displays will be justified.

The re-entry configuration for the SCS consists basically of disabled pitch and yaw channels, roll rate to yaw rate coupling, and BMAG's in a back-up rate configuration. In this configuration, G&N or SCS reference sensors may be selected. Manual attitude capabilities will likely be provided, with the means for displaying the re-entry trajectory to the astronaut currently under study by NAA. Failures during re-entry, or undetected prior to re-entry dictate expedient switch to an alternate mode. Since the implicit information from "flying" maneuvers prior to re-entry is limited in its application to re-entry, and since alternate modes probably will provide a degraded trajectory, it appears essential to verify the re-entry configurations sufficiently prior to re-entry to permit repair, or at least provide expedient decision-making information.

2.1.4 Confidence Testing

Figure 6 lists a representative series of "tests" to be performed for a full confidence test of the SCS system in both static and dynamic situations. Of primary note is the fact that display* information is insufficient during static test situations. Thus, static confidence testing will require display augmentation, particularly for attitude control. The points to be displayed appear to be at least a subset of normally expected diagnostic points, and probably should include the 16 jet driver outputs, in addition to peripheral switching and control monitoring points.

Conversely, the displays, particularly the FDAI, generally provide sufficient information to satisfy the confidence test goals during dynamic operation. It also appears that the FDAI will provide free fault isolation to at least the defective channel, and the function within the channel.

The tests to be performed statically are envisioned as consisting of separate stimulus application to the various signal paths with the channels disabled. The inherent stimulus capability of the G&N system, the BMAG's, the rotation controller, and the translation controller provide the attitude channel stimuli in various combinations. Stimulus of the rate channel is required for full checkout. Such a stimulus is not currently implemented, although partially incorporated. Checkout of re-entry capabilities, in particular, appears to require such a stimulus.

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*Operational displays - no IFTS is considered at this point.

The jet select logic presents a problem in confidence checking in that obtaining the various combinations of translation and attitude commands to provide full checkout in all three channels results in considerable workload, possibly an amount not compatible with earth orbit checkout times. However, the basic common portions of the attitude channels can be checked via a limited exercise of the jet select logic.

"Static" checkout of the TVC circuits is envisioned as the introduction of pitch and yaw error signals, and the observing of gimbal response via the GPI.

The accelerometer and associated electronics currently require a stimulus for static checkout. Results of such a stimulus application could be observed at the ΔV panel.

An additional requirement exists for a built-in "reasonableness" check of display circuits, primarily for operational purposes. M-H is currently studying built in self-test circuitry for displays.

2.1.4.1 Conclusions

The previous discussions have identified a reasonable approach to confidence testing and may be summarized as follows:

1. Provide a static earth orbit confidence checkout capability for the entire SCS, including the following signal paths:

    a. Attitude Channels

    b. Rate Channels

    c. BMAG Sensors

    d. G&N Reference to SCS

    e. Thrust Vector Control Circuits

    f. Accelerometer and Electronics

    g. Engine Fire Circuits

    h. Mode and Re-entry Switching

2. After leaving earth orbit, perform attitude confidence tests implicitly from operational maneuvers.

3. Perform confidence check of ΔV circuits in sufficient depth to verify primary and alternate modes prior to ΔV.

4. Perform confidence check of re-entry circuits in sufficient depth to verify primary and alternate modes (in time to permit repair action) prior to re-entry.

5. Provide a capability to quickly assure that displayed information is reasonable.

These conclusions contain several implications, relative to astronaut task load and hardware implementation.

1. For static checkout, channel stimuli will be required in excess of the inherent stimulation capability. Specifically, rate channel and accelerometer stimulation in the form of D.C. torquer voltages are not currently provided. Measurement and display of several parameters will be required to compensate for the display insufficiency (estimated 16 - 25 points). Appreciable astronaut workload will be encountered in applying the stimuli in the proper combinations during a procedural routine. If earth orbit dynamic control capabilities become reality, the static check capability remains desirable to permit checkout of the system without fuel penalty.

2. Display of the jet driver outputs (16) also appears to be desirable for continuous monitor to detect "failed on" conditions which might go otherwise undetermined at considerable fuel penalty.

3. Display self-tests can be implemented in the form of a "push-to-test" button causing a display reaction recognizable by the astronaut as correct.

2.1.5 Diagnostic Testing

Assuming a failure has been detected by a static confidence test, the logical diagnostic procedure appears to be an extension of the confidence test to include the measurement of as many parameters as necessary to provide the depth of isolation desired, without disturbing the test condition established by the confidence test. Thus the diagnostic test mode in this situation is static, open-loop.

If the failure is detected through operational performance, a diagnosis might be made in a closed-loop dynamic condition. However, actual measurements under closed-loop dynamic conditions present a difficult problem in a servo system. Additionally, an "open" or "hard-over" failure will dictate diagnosis via static test with the channel disabled, unless the hard-over failure is immediately diagnosed. Thus, it is concluded that the primary mode for diagnostic testing is static, open-loop.

Such a condition implies a requirement for static channel stimulation, and for specific test routines requiring astronaut time to establish the test configuration.

Diagnostic testing to provide single module fault isolation appears particularly desirable in the SCS in order to use with advantage the total modular design. Also, the relatively long repair times and the large number of connectors to be disturbed during module replacement strongly suggest such an approach.

With this objective, the number of test points required within the system to provide single module isolation has been estimated at 150 as shown in Figure 7. It is noted that such a set may vary depending upon the amount of correlation one is willing to require of the astronaut, involving both physical and mental activities, and the test sequence chosen. The list of points considered assumes that necessary stimuli are available, and considers only very limited correlation activities. Such an approach appears to provide a reasonable distribution of points, capable of handling both predictable and multiple failures. Test points for the display and auxiliary ECA's were estimated with only very limited information on circuit operation and modularization. However, it is believed that the totals represent sufficiently close estimates for purposes of this study. All the points selected are currently available at the GSE connectors at the front of the panel.

The points chosen were then examined to determine the character of the measurements required to provide isolation, assuming stimuli available at least at the extreme input ranges. The results are shown in Figure 8. The compilation indicates that a major portion of the signals to be examined fall within easily attainable AC (400 cps) and DC amplitude measurements. The notable feature of such signals is the probable variation in level at single points, as the stimuli are varied over representative ranges. The few frequency or pulse measurements involved appear to be amenable to signal conditioning into easily measurable DC levels. Of the 150 points, only a small number (less than 10) are likely to be examined to isolate a single failure.

2.1.5.1 Conclusions

Diagnostic testing of the SCS is reasonably suited to single module fault isolation, particularly within the ECA's. Further, maintenance times coupled with potential introduction of faults through substitution of modules, justify an attempt at single module isolation. Test procedures will likely consist of establishment of static outputs through open-loop stimulation of the various channels. Such stimulation should consist, as a minimum, of a capability to apply representative torquer signals to each of the sensor elements. In particular, such a capability does not currently exist for rate gyros, accelerometer, and attitude gyros in the back-up rate configuration. For fault detection, the application of static stimuli in the required combinations, will necessarily involve astronaut manipulations of several controls, and may involve additional workload on the G&N system. Although a positive requirement has not been established, due to uncertainty of workloads and equipment design, a source of precision AC and DC stimuli at low levels is a reasonable requirement. Given methods for fool-proof insertion, such stimuli, used in lieu of inherent stimuli and self-test torquer voltages, could at least appreciably improve the confidence and diagnostic test capability from the standpoint of workload, efficiency, and versatility.

The range of measurements to be made is generally amenable to direct measurement in most cases, and to simple signal conditioning for the few not easily measurable. The distinguishing feature of the measurement requirement is that the dynamic nature of the system while "flying" makes measurements difficult, generating a probable test requirement for inserting stimuli at varying levels with the system in the static condition, while observing similarly varying outputs at single test points. Such a condition implies that the measurement instrument provided must be capable of selectively observing varying levels. The procedural routines required to establish the static test conditions will involve sufficient manual operations to negate any significant test time decreases realized by "automation".

2.2 Electrical Power System

Other sections describe systems which are primarily electronic in nature. In order to broaden the base of our study, we have also examined the test and maintenance aspects of the propulsion and electric power systems. To illustrate, let us examine the power system in detail (Figure 9).

Three fuel cells generate the power used during the mission until service module separation. Normal missions involve the cells operating continuously with the load distributed across all three. Two radiators supply the necessary cooling. Hydrogen and oxygen supplies are contained in redundant tankage and connected through parallel and series redundant valves and plumbing to permit individual shutdown of each of the cells. Each cell can supply all essential power loads during the mission with the aid of the re-entry batteries for temporary overloads.

The generated power is supplied to redundant busses for distribution (Figure 10). Circuit breakers are supplied to isolate any fuel cell or load from each of these busses. Care is taken to assure continuous power to all essential loads in the event of the loss of either buss. In addition, non-essential loads are collected on a separate buss for easy isolation in the event of a power emergency.

A static inverter is used to convert the DC to 400 cycle three phase power. Each of the three inverters can supply all AC loads. One is used until it fails at which time, it is switched out of the system and the second unit is connected and used. In other words there are two built-in spares. The output of the inverter is applied through relay contacts to redundant busses and the distribution is similar to the DC busses.

Redundant batteries are supplied for re-entry power and a third battery is planned for use after landing. The re-entry batteries serve as a back-up to the post landing battery. Battery chargers are provided to assure full charge at the time of service module separation.

Now consider the power system from a testing aspect. In order to establish whether the power system could perform full load and emergency power functions, the continuous load can be switched from one source of power to another. As an example, a good test would be to apply the load to one fuel cell and monitor the output water flow and generated power. A similar test can be applied to the batteries and the charge rate can also be monitored. The planned command module controls and displays would provide this capability. Similarly, the inverter input and output can be monitored. Checks of the supplies of expendables, temperatures in the various units, and input pressures will supply information with respect to future performance capabilites and hazard conditions. Again, the planned controls and displays supply this capability. Because of the extensive redundancy and the manual controls supplied to switch units in and out, fault isolation can be accomplished most of the time with no other facilities. Where this is not adequate, a multimeter and probing will be required. Therefore, the controls and displays supply full fault isolation requirements for this system with the exception of the multimeter.

A test not included currently and strongly recommended is the measurement of radio frequency noise in the 10kc to 1 megacycle region. Studies sponsored by Aeronautic Systems Division, Air Force Systems Command, show that the measurement of RF noise amplitude and frequency content will often identify malfunctions in electronic and electrical equipment before they are indicated by any degradation in performance. The equipment needed for this test is a tunable filter used to feed a crystal-video receiver and an output meter. A fallout of this measurement is that this test will permit "fault prediction" for all of the electronic and electrical equipment as well as the power system. Since the noise is measured on the power lines, the source of noise can be isolated by selectively switching loads from one power distribution source to another.

Now consider whether maintenance is practical in the power system. The fuel cells, expendables, and radiators are mounted in the service module where access is restricted. However, the busses are behind the circuit breaker panels and these panels are hinged for easy access. The inverters and batteries are located below the command module floor where they can be made accessible. The busses and inverters are therefore likely candidates for maintenance.

Because of the heavy gauge wire that must be used in power distribution, repair actions on these items such as adding insulation where a breakdown has occurred or making a connection where a break has occurred do not require delicate or precision work and therefore are considered practical operations.

The present inverters weigh 36 pounds each and are not modularized. Three inverters account for 108 pounds. Each of these inverters could be packaged in four modules; one module containing the power handling portion for each phase (three modules being identical) and the fourth module containing the modulating oscillator, amplifiers, and feedback circuits. Two built-in inverters and two spare modules (one of each type), if module replacement were employed, would provide higher mission success probability than currently obtainable with the three inverters. Moreover, this module replacement approach requires approximately 15 pounds less weight. Further gains in mission success probability could be obtained by permitting component replacement of the power transistors (within each phase power module) since they are the most probable source of failure and are physically large by comparison with normal transistors.

Summary

The planned controls and displays will provide extensive testing capability.

Conductive RF noise measurement should be studied for use in augmenting presently planned measurements.

The inverter case that thas been cited shows that in heavy systems where extensive redundancy is used and repair is feasible, less redundancy and sparing could provide higher mission success probability for less weight.

The power system shows maintenance opportunities exist where none are planned. Similarly, there are many other systems where no maintenance is planned, and it is recommended that these also be examined for maintenance opportunities.

2.3 Communications and Data Subsystem

2.3.1 System Description

The communications and data (C&D) subsystem provides the realization of the Apollo functional requirements for voice, telemetry, tracking, ranging, recovery information, and data recording with playback. The equipments divide fairly naturally into near-earth and deep-space equipments, although certain equipments are common to both and hence are switched to the appropriate mode of operation at any given time. Figures 12 and 13, attached, illustrate the two systems as they are currently conceived.

Characteristic of the near-earth equipment is the fact that individual transmission facilities are provided for each service required; that is, voice service is provided over one particular transmitter and receiver combination, telemetry over another, and tracking over another. All have their own modulation techniques and frequency ranges. In contrast the deep-space equipment is characterized by the fact that all the aforementioned services are provided through a single transmitter and receiver combination, using either phase or frequency modulation on an S-band carrier frequency. The equipment common to both near-earth and deep-space phases is input equipment and includes the audio center, the PCM telemetry with its associated signal conditioning, and the data storage recorder.

At the present time several equipments which belong to the total communications and data subsystem are in an undefined state, either because of recent changes in concept or because certain necessary interfaces are not established. Thus, for the present, the equipments listed in Figure 14, attached, will not be considered as parts of the C&D subsystem although, in fact, they are.

Certain physical characteristics of the equipments that are under consideration are listed in Figure 15, attached. It is very likely that with the addition of the aforementioned undefined equipment the total weight of the system will increase to approximately 300 pounds and the total number of modules to approximately 130. The total number of module types listed is misleading because of the type of duplication possible. For example, in the signal conditioners there are 22 fixed dc amplifier modules, identical in general design but differing in gain. For this reason, one cannot conclude that the sparing problem is helped any by this type of module similarity. In effect, there are 22 different signal conditioners from the viewpoint of sparing, although inherently the module types are alike. Several other similar situations prevail, but are of lesser concern because fewer modules are involved. Another point to note on the listing of physical characteristics is that assigned equipment reliabilities are, roughly speaking, an order of magnitude beyond the current predicted reliabilities of the equipments. It is obvious that assigned reliabilities will be met only by the provision of a maintenance capability in the equipments as currently designed.

A sketch of the equipment placement is shown in Figure 16, attached. The view is what one would see looking toward the right leg and foot while reclining in the right-hand side astronaut couch. To a degree the equipment bay is built in the fashion of a chest of drawers, although the drawers in this case are bolted in. The equipment packages (drawers) are frames with modules mounted intimately together and held fast with thru-bolts. The average time to replace a module -- i.e., remove drawer, remove module, replace module, replace drawer -- is estimated at about 40 minutes for a weightless, shirtsleeve environment. The S-band transponder is the only piece of equipment which is physically redundant, and because it is, the time required to replace this unit is somewhat reduced. Alternate modes which exist for the equipments under consideration are listed in Figure 17, attached.

2.3.2 System Testing

2.3.2.1 Confidence Testing

Confidence testing is the result of the necessity to make equipment-dependent operational decisions. Figure 18, attached, shows an estimate of when during the nominal mission profile particular equipments would be checked out. In essence this chart says what one might intuitively feel -- that during and before deep-space operations the deep-space equipment should be checked, and that during and before near-earth operations, the near-earth equipment should be checked. It is assumed that confidence tests will be performed sufficiently in advance of the time when equipment becomes operational to allow time for maintenance or initiation of an alternate mode.

In the C&D subsystem there are four classes of equipments which must be checked: transmitters, receivers, signal processors (i.e., filters, amplifiers, etc.), and time-dependent common equipment (PCM). Figure 19, attached, lists two estimates of confidence information requirements for these four classes of equipments. The first column lists the information considered sufficient for confidence purposes. The second column lists the information considered both necessary and practical to provide.

Confidence tests will be tests which measure, perhaps grossly, the overall performance of a functional unit (i.e., a group of equipments such as the HF equipments, the VHF-AM equipments, etc.). As an example, a confidence test of the VHF-AM voice equipments could consist of asking a question of the ground personnel and receiving a recognizable reply. This simple test would verify, at least grossly, the integrity of the VHF-AM equipments, including transmitter, receiver, filters, audio amplifier, microphone, switches, antenna, and wiring. In fact, the information required by column two of Figure 19 will have been brought forth, if the test gives a positive result. It should also be observed that, in this example, use is made of "free"* information available from the ground, in accordance with a previously established principle.

2.3.2.2 Diagnostic Testing

A malfunction in any equipment will be detected whenever operability is lost, or a confidence check fails, or whenever the ground determines that trouble exists and informs the spacecraft. The purpose of diagnostic testing is to locate the source of a malfunction to some predetermined degree of isolation. Since the C&D equipments are using modular construction, a module is the selected design level of maintainability. Therefore, it is the goal of diagnostic testing to localize a malfunction to a particular module. In most cases this goal is achieved. In a few special cases single module isolation is not a judicious pursuit.

*Free - not requiring spacecraft display.

Malfunction localization through diagnostic testing requires that predetermined operational procedures be followed whenever a malfunction occurs. These procedures will specify how test conditions are to be established, what information is derived from tests, and how the total amount of information available is to be correlated for diagnostic purposes. This set of procedures might take the form of an illustrated trouble-shooting booklet.

Diagnostic testing in modular communications equipment usually resolves into: 1) establishing test conditions (i.e., putting all switches in correct positions, etc.), 2) checking power supply voltages, and 3) making signal-flow tests. The number of tests made depends strongly on the number of functions performed by the equipment. Thus single-function modules may undergo no more than one or two tests, whereas multi-function modules may require several tests to verify availability of all functions.

In selecting diagnostic test points, a trade-off immediately arises between the number of test points and the amount of information correlation necessary. A large number of test points permits diagnosis with little information correlation. Similarly, diagnosis with a small number of test points requires considerable information correlation. A matter of judgment is involved and the judgment used in this report has been that enough test points should be provided so that only a limited amount of correlation is necessary -- an amount readily expected of anyone knowing only the rudimentary principles of communications equipment.

Establishing test conditions for diagnostic purposes may involve the introduction of externally provided stimuli. For example, a test of a voice transmitter would probably require that an audio modulating signal be externally introduced into the transmitter's modulator. Similarly, to test a voice receiver, it would probably be necessary to externally introduce into the receiver an audio-modulated carrier signal. Stimuli considered desirable but not currently available for C&D equipment testing are modulated carriers for the HF, VHF-AM, and S-band receivers. The S-band carrier would be used for checkout in earth orbit. The HF and VHF-AM carriers would be used for checkout prior to re-entry. While it is not anticipated that these stimuli would be required at other times, having a positive test capability independent of ground-supplied signals still appears desirable.

2.3.2.3 Testing of PCM Equipment

Figure 20, attached, shows a very simplified block diagram of the PCM data encoding system which produces the modulating signal for the telemetry transmitter, in addition to timing signals used for other purposes. One notes immediately that the programmer interacts with all of the other functional blocks.

To test this PCM system for confidence purposes, it is necessary to test the performance of each of the data channels, both analog and digital. This means introducing known inputs on the channels and observing the NRZ-PCM digital output. From the viewpoint of one making an external test, rapid switching and fixed display of the coded output is required.

If a given channel or group of channels fails a confidence test, diagnostic tests are required to locate the source of the malfunction. Diagnostic testing implies checking the performance of each of the blocks shown. This must be done by verifying that the programmer issues, at the correct time, the correct instructions and that the functions of the various blocks which occur in response to the instructions are indeed correct.

The problem of instrumenting such tests with external test gear are formidable. This is principally because of the time-dependence of the measurements, and because very rapid switching must be verified. One does not diagnose faults in such a system with a simple voltmeter or comparator. Rather, logic and switching and a certain amount of programming are necessary. In short, to test this kind of circuitry requires kindred circuits. It has been judged that the presently proposed self-test circuitry for this equipment is the logical means for implementing both confidence and diagnostic tests, provided the present modular construction concept is adhered to.

Associated with the PCM equipment are the signal conditioners for the analog input signals which require them. This group of about 125 separate signal conditioners plus power supplies (mounted in 44 modules) represents a difficult checkout problem because valid testing of a single signal conditioner requires that both the input and output be measured and that the two measurements be correlated. Thus over 250 measurements plus calibrated stimuli are required to check out the signal conditioners alone. In this case it is judged that single module isolation is not a prudent objective to pursue. If the ground notes that particular telemetry channels are malfunctioning, trouble shooting is probably best left to the ground which can inform the spacecraft if particular channels should be repatched, deleted, etc. It is inherent in this concept that critical telemetry signals may be telemetered over more than a single channel.

2.3.3 Conclusion

There are two classes of tests offered to any proposed test equipment by the C&D subsystem: 1) those tests requiring a complex instrumentation tailored specifically to individual circuit designs, and 2) those tests which require only simple time-independent ac or dc voltage measurements.

The first group of tests are those associated with the PCM equipment in particular, and digital data handling equipment in general. Thus the data storage recorder tests would have been included in this group but because there exist no in-flight maintenance requirements for this equipment, its testing has been dismissed. The difficulty in testing the PCM equipment has already been pointed out and it is judged that, as long as modular design is adhered to, the best solution to testing this equipment is to have built-in self-test circuitry as opposed to having an external testing capability.

The second group of tests referred to are the voltage measurements one would encounter in checking out transmitting and receiving equipments. In this context it has been assumed that rf test voltages are signal conditioned to dc. There are two reasons for this: 1) the number of rf measurements is small, and 2) it is desirable not to measure rf voltages with a central test system. Thus, the voltages to be tested can be measured by a general purpose voltmeter and/or comparators. Since there are no stringent requirements for specification-tight accuracy, measurement instrumentation would not be difficult from this viewpoint.

Figure 21, attached, gives a list of test points which, under non-failure assumptions on the switches, wiring, and passive equipment, would be sufficient with few exceptions for confidence and diagnostic tests to the modular level. However, a reasonable amount of information correlation and subsystem knowledge would be required to augment these test points. Another slightly more generous list of test points is shown in Figure 22, attached, which requires only simple correlation of information for confidence and diagnostic purposes. In many cases, an audio indication heard from a headset is a valid test which can do away with test points. The total number of test points is not great. This is due chiefly to the fact that the C&D modules are large.

There exist at least two possible sources of perturbation for the conclusions presented: 1) the undefined equipment mentioned in the system descritpion, and 2) the so-called unified frequency approach to the Apollo communications requirements.* Neither of these sources of possible perturbation would be expected to significantly alter the number or type of test points to be served from the C&D subsystem.

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*This approach would make the Apollo spacecraft near-earth and deep-space equipments virtually identical. The greatest changes would occur in the ground facilities.

2.4 The Guidance and Navigation System

The major parts of the G&N system are shown in Figure 23. The scanning telescope (SCT) has a wide field of view for use by the astronaut in star finding and in short range tracking. It is a single line of sight instrument having two degrees of freedom with respect to the spacecraft. The sextant (SXT) is a precision, high-magnification, narrow-field navigation instrument having two lines-of-sight. During navigation measurements, the scanning telescope (SCT) is used to search for and identify the star and to bring the star into the center of the field of view. The sextant (SXT) with its high magnification, but narrow field of view, is then used to make the actual measurement. The IMU is the primary inertial sensing device. It is a three gimbal system. Three accelerometers are mounted on the innermost gimbal which is held nonrotating with respect to inertial space by the action of error signals from three gyroscopes. The signals from the accelerometers measure the vehicle acceleration in inertial coordinates, and the three gimbal-angles correspond to the attitude of the spacecraft relative to the inertial coordinates. For initial alignment, the stabilized-member gyros can be torqued from the computer to precess the stable member.

The coupling and display units (CDU) are used to communicate angular information between the IMU, the computer, and the astronaut, as well as to generate the attitude error signal for the SCS system.

The Apollo guidance computer (AGC) is the central processor of the guidance and navigation system.

The display and control system (D & C) is the interface between the astronaut and the guidance and navigation system. The D & C contains the computer keyboard and display, the controls necessary for operation of the SCT and SXT, and the controls necessary for alignment of the IMU. The eightball (FDAI) is an important display for monitoring the G&N performance, but this instrument is considered part of the SCS system.

The power servo assembly (PSA) contains the power supplies for the IMU and Optics, as well as all the servo systems that are used in conjunction with the IMU and Optics. The PSA equipment consists of three functional parts: 1) the power supplies and servos that are used to control the temperature of the IMU, 2) the power supplies and servos that are used for IMU operation, and 3) the power supplies and servos that are used for optics operation.

There is one more important part of the G & N system that is not shown in Figure 23. This is the map and film viewer. This viewer is provided in the G & N system to store maps of the stars, moon, and earth, but by the addition of more film, can be used to store detailed astronaut instructions, circuit diagrams, and other detailed written material.

Virtually all of the G & N equipment is located in the area called the lower equipment bay (Figure 24). During stress periods when the astronauts are in their couches, the G & N equipment is inaccessible. The feet of the middle astronaut are positioned just below the eyepieces of the telescope and sextant. To gain access to the G & N equipment, the middle couch is removed, folded up, and stored under the pilot's couch on the left. A part of the display and control equipment is duplicated on the main panel display over the pilot position. This display consists of keyboard and computer display plus some warning lights that indicate gross malfunctioning of the G & N system.

The major physical characteristics of the G & N system are shown in figure 25. The characteristics are weight, power, MTBF, duty cycle, whether or not the subsystems are repairable and, if so, the number of working modules in the subsystem. The data for the weight, power, and duty cycle was obtained from the 15 May 1963 weight and balance report issued by MIT/IL. The MTBF figures are estimated reliability which came from an MIT/IL report R395, issued in February 1963.

The total weight of the G & N system is 475 lbs. This figure includes a weight allocation of 53 lbs for spare modules. The spares weight considers 37 lbs. of spares for the AGC, 9 lbs. of spares for the PSA equipment, one spare CDU (3.5 lbs.), plus a tray (4 lbs.) for housing some of the AGC spares. This spares tray only holds 11 lbs. of spares so the additional mountings, as well as additional cabin space will be needed to hold the rest of the spare modules.

No power consumption is shown for the PSA equipment. The power supplies and servos certainly consume power but as mentioned earlier, part of the PSA is used in conjunction with the IMU and part with the Optics. The power consumed with these parts is indicated under IMU and Optics respectively.

The 15% duty cycle indicated for the IMU is valid only for the operational parts of the IMU. The temperature control system for the IMU is, and must be, operated continuously.

The duty cycle indicated for the AGC is a stab at a moving target. The 33% figure comes from the weight and balance report. Other MIT reports indicate a duty cycle ranging from 10% to 100%. The trade-offs in this area are not only the desire to conserve electrical energy by leaving the computer off when there is no work to be done, but also to increase the AGC reliability by leaving it off. The reliability argument is that there are fewer failures with power-off than with power-on, and provided that the turn-on, turn-off stresses are not severe in themselves.

There is virtually no data on the ratio of failures in power-off to failures in power-on, and the fudge-factor to be used in reliability estaimates is a matter of considerable speculation. This area is further confused by the fact that the AGC is never completely off. The AGC contains the master clock for the entire CM and this clock cannot be turned off. "Off", for the AGC, means that for 990 ms out of every second the entire computer is truly off except for the oscillator and the frequency dividers hanging on the oscillator. However, for 10 ms out of every second the computer is turned on and during this 10 ms all circuits are functioning, and functioning at full speed.

Within the G & N system, the only parts that are being constructed in replaceable modules are the PSA and AGC. The IMU and Optics are not repairable in flight. The navigation base is merely a structural member for rigid mounting of the IMU and Optics. While it may be possible in principle to consider an astronaut repairing a broken wire, in practice, the problems are formidable. The problems are accessibility, soldering in a vacuum, or pure oxygen, as well as the problem of identifying and finding the broken wire in the first place. Wiring is not considered repairable.

The main part of the DISPLAY is completely duplicated. There is a computer keyboard and display on the main display panel as well as in the lower bay. In addition, some of the self-checking features that are built into the G & N system cause duplicate alarm lights to go on in both the main display panel and the lower bay.

The main reason for the duplication of the display equipment is functional. During the periods that the astronauts must be in the couches, the lower equipment bay is inaccessible, and the second display is considered necessary. Whatever the original reasons were, this is redundant equipment design. Given that one of the displays fail, the astronaut should be able to make do with the remaining display.

The PSA, the AGC, and the CDU's are constructed in replaceable modules, and in the current reliability estimates, sparing and in-flight maintenance are being heavily relied upon to achieve the reliability goals that have been set for these parts of the G & N system.

The next two sections consider the problems of fault detection (confidence tests) and fault isolation (diagnostic tests) in the PSA and AGC respectively.

First, some comments are necessary about confidence testing of the complete G & N system. Suppose that a number of sextant sightings are made and the AGC calculates, as a result of these sightings, that a mid-course correction should be made in say 1/2 hour at a specified attitude and for a specified duration. Assume further that during these sightings all equipment functioned correctly -- as far as the astronaut can tell. The astronaut may apply certain weak reasonableness checks on the ΔV calculation, but for all practical purposes, there is no on-board capability for verifying that this ΔV is "correct". The capability does exist on the ground at IMCC and it is presumed that all mid-course calculations are verified with the ground before any engines are fired. The ground, in fact, may be capable of performing "better" ΔV calculations than is possible on board the spacecraft during some phases of the mission. The point to be established here is that the main confidence test for the performance of the complete G & N system is on the ground, and not on the spacecraft.

2.4.1 Power Servo Assembly

The PSA equipment can logically be broken down into three different parts. They are:

a. IMU Temperature Control
b. IMU Servos
c. Optics Servos

Separate power supplies are used for these different parts of the PSA so that the IMU servos and/or the optics may be turned off when they are not needed. The temperature control for the IMU is never turned off.

There are nine servo loops used in conjunction with the IMU (figure 26). Three for the accelerometers, three for the gyros and three for the gimbal angle measurement. The modes for these servo loops must be controlled to permit the coarse and fine alignment of the IMU as well as the operational use of the IMU during thrust periods.

The greatest versatility in the mode-switching exists in the gimbal angle measurement loops. A dominant part of the angle measurement loop is the CDU. There are three CDU's used in conjunction with the IMU, one for each angle (two more CDU's are used with the optics).

The main element of the CDU is a shaft (actually shafts with gearing). The current position of the shaft is displayed for the astronaut on a dial and changes in the shaft position are detected and sent to the AGC computer. The position of the shaft may be controlled mechanically by the astronaut by means of a knurled wheel on the shaft, or electrically by means of a switch, amplifier, and servo motor on the shaft. The computer may also control the CDU shaft position by means of a digital-to-analog converter and the same amplifier and servo.

Two resolvers on the CDU shaft compare the CDU position with the angular position of corresponding gimbal angle. A third resolver is needed in each CDU to initially "zero" the CDU position. The comparison of the resolvers results in an error signal. The use of the error signal depends upon the mode of the IMU.

During coarse alignment, the error signal is sent to the gimbal servo amplifier and in this manner the position of the IMU is forced to follow the position of the CDU.

During fine alignment the error signal is sent to the amplifier and servo motor that controls the CDU shaft position, and hence the CDU is forced to track the IMU position.

During propulsion periods, the IMU measures the current attitude of the vehicle in inertial space, the CDU is set by the AGC computer to the attitude that the vehicle should have, and the error signal is sent to the SCS system for correction.

This is an appreciable number of modes, and there are only three modules in each angle servo-loop plus the CDU and the IMU. All of these modes are normally used by the astronaut during the alignment of the IMU. If a hard failure has occurred since the last time the IMU has been used, the astronaut can first detect the failure by the absence of the normal response, and second, isolate the failure fairly well merely by noting the angle that is experiencing the difficulty and the step in the alignment procedure in which the trouble is first found.

To provide further assistance in fault isolation, it is desirable to provide a capability for measuring some of the signals flowing between modules. By means of 39 measurement points of this type in the PSA equipment, the fault may be isolated to within three modules.

Fault detection and isolation in the optics portion of the PSA can be performed in an analogous manner. The accelerometer loops pose different problems. There is only one mode in the accelerometer loops. By monitoring the outputs of the PIP's excessive precession may be detected and alarms given. This alarm will detect one class of failure but it is of no use if the amplifier just ahead of the monitoring point goes dead. This second class of failure can be detected during propulsion phases by reasonableness checks on the velocity vector. During non-propulsion periods, normal astronaut movement plus an idiosyncrasy of the accelerometer loop should cause the least significant bit in the velocity vector to vary. Absence of this variation is an indication of malfunction.

There are power supplies in the PSA equipment, and a failure in a power supply can result in confusing performance of the IMU and/or Optics. It is assumed that the first task that is performed in any fault isolation procedure is to check the power supplies. Fourteen additional measurement points are needed for this purpose.

At this point, some of the assumptions that are implicit in the above discussion should be noted. They are:

a. Only hard failures are considered.

b. The astronaut is permitted to manipulate the modes at will to detect and to isolate the fault. Specifically, he is permitted to destroy the current alignment of the IMU.

c. The astronaut has the time to do this manipulation.

By considering only hard failures, the much more difficult problems associated with gradual degradation in performance are ignored. In particular, there is very limited (if any) on-board facilities for detecting when the drift of the IMU becomes "excessive". In addition, the fault isolation procedure briefly described above is of no use if the hunting in a servo loop gradually becomes excessive, or if a servo loop breaks into violent oscillation.

During the earth parking orbit, it is highly desirable not to destroy the alignment of the IMU. There may be a problem in performing a confidence test on the PSA and IMU during earth orbit because of this constraint. In particular, the inner gimbal of the IMU is to be aligned normal to the orbital plane to achieve greatest accuracy. If this alignment is very close, then one gimbal angle and one gyro may not be exercised sufficiently while in the parking orbit to verify that they are capable of functioning.

During the propulsion periods, the astronaut does not have any time for fault isolation. The only item of interest is quick fault detection. Some fail-safe features are built into the G&N system by continuously monitoring the nulls on the accelerometer, the gyro, and attitude servo loops. If the error voltages appearing on these nulls become excessive for an appreciable period to time, then a failure light is put on. If one of the accelerometer loops fails, the velocity integration within the computer is also terminated. If the gyro or attitude loops fail, the G&N system disconnects itself from the SCS system.

In summary, the various modes provided within the PSA equipment for normal operation also provides good confidence tests on the PSA performance. A failure may be quickly isolated to a particular servo-loop by the displays. By providing the capability for measuring some of the signals between modules in each servo-loop, the fault may be isolated to within three modules. Thirty-nine measurement points are needed for this purpose plus 14 additional measurement points for checking the PSA power supplies.

There may be some problems in performing a confidence test in earth parking orbit without destroying the alignment.

2.4.2 Apollo Guidance Computer (AGC)

The Apollo Guidance Computer is a stored program, general-purpose computer. There are a number of small, general-purpose, scientific computers on the commercial market that are similar to it in capability. The unique feature of AGC is that it is being designed to satisfy the severe weight, volume, and power constraints of Apollo, and the input/output abilities of AGC are tailored to the needs of the CM. The main computation features of AGC are summarized in Figures 27 and 28. A very readable, but yet detailed, description of the logical performance of AGC may be found in MIT/IL report R393.

There are four major causes of errors made by digital computers. They are:

1. Logical and Clerical Program Errors
2. Operator Mistakes
3. Transient Failures
4. Hard Failures

2.4.2.1 Program Errors

For Apollo, programming errors are not to be tolerated. Prior to lift-off extensive simulation and program checking will be performed to identify and remove all programming mistakes. Some mistakes may sneak by, but the number is to be minimized by all available means.

2.4.2.2 Operator Errors

The astronaut has the ability to enter data into the computer, and errors in the entry of this data and/or incorrect data will cause mistakes. Astronaut training plus the interlock scheme proposed for keyboard entry should minimize the possibility of operator error.

2.4.2.3 Transient Failures

Transients are here defined to be the occasional dropping of a bit or the creation of a bit. By definition the malfunction cannot be repeated, or cannot be repeated easily.

The importance of transient failures is dependent upon the consequences of a single failure, and the number of transient failures that might be expected.

The consequences of a single failure depend entirely upon which bit was destroyed and when. If the bit was the least significant bit of a number, the error may be of no consequence. If it was the most significant bit, reasonableness checks might pick it up. The most disastrous consequences of a transient may occur if the error is made in the program, or in data that is treated as part of the program. In the AGC, it is possible that a single transient may result in the destruction of the entire contents of the temporary memory. This is a possibility. The probability is impossible to estimate.

It is extremely hard to detect when a transient failure occurs, and hence also difficult to get any estimate of the number of transients that occur. Some data on transients has been gathered in the SAGE system. The SAGE computer facility consists of two computers each solving the same problem and each checking the detailed performance of the other. If a mismatch occurs, the computer traps, the error is logged, and another attempt is then made to execute the instruction correctly. The data from SAGE indicates that transients are a factor of from 5 to 10 more frequent than hard failures. This data is not necessarily applicable to AGC, but it is the best data on transients, known to the author.

Detection of the occurence of transient failures is usually difficult, but more significantly detection is useless in AGC unless automatic means for correction are also available. The parity detection circuit in AGC can detect some transient failures, but even if the detection is made, the only satisfactory procedure is to correct the parity bit, light the alarm light, and then proceed as if no error had even been detected. It is not possible in the AGC to attempt a second execution of the same instruction to try to get better results. The computer can be designed to stop upon detection of a single parity failure, but this is equivalent to stating that the consequences of a single failure are worse than the consequences of no computations at all. In a guidance computer, this is definitely not correct.

In short, transient failures are a risk inherent in the design of the AGC. The magnitude of the risk is hard to assess.

2.4.2.4 Hard Failures

Hard failures, in contrast to transients, are relatively easy to detect. The AGC computer is being designed with limited self-checking abilities (figure 29). The most useful are parity on the memory, a test program, and input/output echo checks.

A continuous stream of parity failures indicates that the computer memories, or the circuitry immediately adjacent to the memory is malfunctioning. Parity provides good fault detection on the computer memory, but it provides little fault isolation. Given a hard parity failure, the suspect areas are the memory cores (14 modules), the memory drivers and sense amplifiers (about 4 of 5 modules), and the memory buffer register, bus, parity register, and parity check circuit (about 18 to 20 modules).

A test program is a program specifically designed to exercise all the capabilities of the computer. If this program runs successfully, or merely runs at all, then there can be high confidence that no failures exist. A malfunction will cause the test program to stop, or to produce meaningless results. The test program is useful for detecting malfunctions but is little or no value in performing fault isolation.

Echo checking is the ability to connect computer output to the computer input, and the functioning of the input/output sections of the computer connections must be physically rearranged before echo checks may be made, and hence echo checks are of little value in fault detection. The ability to perform fault isolation in the input/output portions of the computer, is limited only by the ingenuity of the programmer and by the length of the test program that is tolerated.

There are other self-checking features in the AGC4 computer. "Inactivity" and "scaler fail" monitor the basic oscillator and the long frequency divider that hangs on the oscillator. TCA trap and RUPTLOCK will detect gross malfunctions of the computer. The power supplies are monitored.

THE PRESENT COMPUTER SELF-CHECKING FEATURES ARE OF NO VALUE IN DOING FAULT ISOLATION IN THE "CORE" OF THE COMPUTER. THE "CORE" CONTAINS ABOUT 90% OF THE LOGIC CIRCUITS AND 75% OF THE WEIGHT OF THE COMPUTER.

Four approaches to the maintenance problem of the computer are being examined. One approach is to add a maintenance console to the AGC so that the astronaut may force the computer to execute primitive actions and then observe the results. Some weight must be added to the AGC for the console, but this approach also requires the astronaut to be a highly skilled maintenance man and that he can concentrate upon the task of fixing the computer for a fairly sustained period of time (1/2 hour to 2 hours). Even with the console and extensive training, the ability to perform fault isolation to 3 of 4 modules is uncertain.

The second approach is to ignore the fault isolation problem and to examine cut-and-try replacement of all the modules that might be faulty. This approach is excessively time consuming (1/2 hour to 5 hours) and has other serious drawbacks. One difficulty is that massive replacement of modules is going to exercise the connectors in the AGC, and connectors are not as reliable as would be desired. If, in the course of replacement, a connector contact is damaged, the maintenance action induces a second trouble in the computer. With two troubles in the computer, the astronauts' chances of clearing these troubles by cut-and-try techniques are nil.

The third solution is to again ignore the fault isolation problem and to examine cut-and-try replacement of complete drawers. The entire AGC is contained in three drawers, and the repair time should be short (about 15 to 30 minutes) provided that the middle couch is stored away and the astronaut has access to the lower equipment bay. There is some chance of inducing additional troubles in the drawer connectors, but compared to the second approach, the risk is much smaller and is considered acceptable. When complete drawers are spared, there is a benefit that spares are carried for the module connectors and for the drawer wiring. The penalty of this approach is weight. Virtually a second computer is being carried on a spare. The only parts omitted are the back-panel wiring. Weight-wise, sparing on a per-drawer basis is equivalent to redundancy without the benefits of redundancy. There is no protection against transient failures nor against a failure during a critical time (major propulsion periods and reentry).

The fourth approach is to design a redundant computer for use as AGC. A dual computer system will provide fault detection (transients as well as hard failures), but as described earlier, automatic fault correction is also needed. This approach contains a weight and power penalty, but the penalty in weight is not as great as is indicated by a quick judgment.

In summary, maintenance of the AGC is an unsolved problem at the present time, and it is a difficult problem to solve. A redundant computer design is the recommended solution.

The maintenance tasks associated with the AGC are different from the maintenance tasks associated with other systems in the CM. The instruments and/or techniques that are developed for the maintenance of the AGC will be unique to the AGC, and will not, in general, be applicable to the other systems.

3.0 Recommendations

The first part of this section will consider the merits of in-flight maintenance vs. redundancy for some of the spacecraft systems. The second part will consider various means of implementing the confidence and diagnostic tests which have been described in the previous system discussions.

3.1 Redundancy vs. In-Flight Maintenance

In the system by system discussions, it was pointed out that, except in the case of the electrical power system, the present estimated reliabilities fall short of the design goals. It was also pointed out that the electrical power system is the only system which has made extensive use of redundancy to increase the inherent system reliability. In the other systems in-flight maintenance and sparing with its associated test equipment has been the primary means of achieving the assigned system reliability. In view of the many maintenance problems that have been pointed out in the computer and PCM areas, one might well ask if in-flight maintenance should be the accepted means of increasing the reliability of these two systems.

To fully answer the question of whether to use redundancy or in-flight maintenance for a particular system, a trade-off study should be made on a modular and non-modular design of the system. Although this was not done for any of the systems described previously, a comparison was made between two similar systems built by the same contractor. The systems considered were the two PCM systems built by Radiation, Inc. One system is presently being used aboard the Telstar satellite, and the other is being built for use aboard the Apollo spacecraft.

Some of the characteristics of these two systems are shown in Figure 30. The item of particular interest in this figure is the breakdown of electrical and mechanical component weights for the two systems. In the Telstar system, there is a weight breakdown of seven pounds of electrical components and one pound of mechanical components. In the Apollo spacecraft system, there is a weight breakdown of 14 pounds of electrical components and 36 pounds of mechanical components. Thus, we have a factor of 18 to 1 in the difference between the weight ratios of electrical to mechanical for the two systems. Although it cannot be stated conclusively that this penalty is strictly due to modularization of the PCM system, a quick look at the two systems has revealed no other apparent reason. It should be pointed out that a sub module used by Radiation, Inc. in these systems is considered to be totally electrical components.

Figure 31 illustrates the mechanical construction of the two systems. As may be seen from the figure, each module in the Apollo spacecraft system* is entirely encased by metal; whereas, in the Telstar system the modules are placed side by side and the entire unit is inserted into a metal container. Although the Apollo spacecraft PCM system as pictured in Figure 31 can be mounted directly to the spacecraft, the Telstar PCM system must be mounted with at least one inch of echo foam all around it, whose weight is approximately one pound. This weight was not included in the weight shown in Figure 30.

In the area of reliability, two items are of particular interest. The first is that, using the present Apollo spacecraft PCM reliability estimate, two such units would be sufficient to meet the specified reliability. The second is that the Apollo PCM system reliability cannot exceed 100,000 hours MTBF due to estimated connector reliability alone.

In view of:

1. the penalties which may exist in terms of weight and reliability for insisting on modularization of the PCM system,

2. the difficulties which exist in diagnosing the PCM system to a modular level, and

3. the similarity between circuitry and diagnostic problems in the guidance computer and the PCM system,

the following recommendation is made.

Review the basic system design concepts of at least the PCM and computer systems to insure that in-flight maintenance and sparing is the best approach to meeting the assigned reliabilities.

The scope of this review should include such factors as reliability, transients, weight, and power. In the reliability calculations, consideration must be given to whether the system is available for maintenance action and if sufficient time exists for this maintenance action.

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*The Apollo spacecraft system consists of two similar equipment packages. Only one is illustrated here.

3.2 Implementation of Confidence and Diagnostic Tests

Although Section 3.1 suggests a review of the design concepts for the PCM and Computer systems, this Section includes a discussion regarding test implementation, insofar as can be recommended, for the same systems, given the present design concepts.

3.2.1 Unique Problems

In Section 2.0, it was noted that the computer and the PCM systems require unique types of measurements. Both systems consist of digital circuitry and require intricate timing interfaces between the test equipment and the prime system. Further, the signals existing in these systems are in the form of pulses of μsec or msec duration, requiring the test system to sample a pulse and hold the result for display purposes. This is in contrast to monitoring and displaying the result, as may be done, for instance with a fixed level DC voltage.

In the computer at present, there is virtually no ability to perform fault isolation. Four possible approaches to this problem are discussed in Section 2.4.2. Two of these approaches (a maintenance console and cut-and-try replacement of modules) are unsatisfactory. Although cut-and-try replacement of drawers in the computer is an acceptable solution to the maintenance problem, this solution incurs the weight penalty of redundancy with none of the advantages of redundancy. A redundant computer design is the only solution that is recommended.

PCM diagnosis has been partially solved by incorporating self-test circuitry on a modular level. However, the system is still only approximately 50% self-checked, and in the programmer (the core of the system) there are five modules which contain no self-test circuitry. Further, the complexity of the self-test circuitry in terms of number of components varies from a "small fraction" of, to "greater than" the complexity of the circuitry being tested. The "greater than" case is a unique situation, and the average percentage of test circuitry probably lies somewhere between 10 to 25 percent for those modules that are checked. These facts are not meant as a criticism of the present PCM design, but rather to indicate the cost of having to test on a modular level for such a system.

Even in view of the weight penalties of the self-test circuitry, the authors feel that this is the best approach to testing the PCM system. Therefore, under the assumption that the PCM system design will remain modular in construction, the outputs of the presently incorporated self-test circuitry are included in the tabulation of test points to be handled by any proposed integrated in-flight test system.

Another unique problem is that stimuli are required for the SCS. It is recommended that the stimuli noted in Section 2.1.5.1 be provided.

3.2.2 Test Points to be Served by Integrated In-Flight Checkout System

Figure 32 contains a summary of the test points that have been suggested in the system by system descriptions. These test points are additional to the displays presently provided. The measurements on these points consist of fixed level, bi-level, and variable level AC and DC voltages.

As shown in the tabulation of points, there is a total of 34 confidence test points and 212 diagnostic test points. It should be recalled that the ground has been assumed to be the primary source of confidence checks on the G & N System, and in addition, the coupling display units and the tests other than those already suggested are required. The total of four diagnostic test points listed for the Apollo guidance computer is not the number needed to diagnose the computer, but is the total of the type that could easily be handled by an integrated in-flight checkout system. If the computer were to employ self-test circuitry similar to that used in the PCM system, the outputs from these circuits would be in addition to the four shown here. The 13 outputs from the PCM self-check circuitry are included in the 26 diagnostic points listed for the Communications System.

3.2.2.1 Alternate Implementation Methods

Figure 33 lists several alternate means of implementing the measurements summarized in Figure 32. Although the list is not exhaustive, it covers what are considered as reasonable approaches to implementation. To help visualize the various proposed techniques, illustrations are given in Figures 34 and 35.

The PCM-Computer configuration depicted in Figure 34 is comprised of the present PCM and computer systems and not an additional PCM and Computer system. In this configuration, the PCM would serve as the information gathering device and the computer would serve as the information processing device. The darkened portions of the PCM and computer blocks represent essential additions to the present equipment and entail minor alterations to the present designs and small weight penalties. The addition* to the PCM system would consist of the circuitry required to accept requests from the computer for information

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*Additional subcommutation circuitry may have to be added if the measurements are not presently being telemetered to the ground.

on a particular channel. Thus it would consist of a channel address matching register and the circuitry for gating the data from the selected channel to the computer. The addition to the computer would consist of the required additional memory to store the accepted levels and tolerances for the various measurements and for the program to process these tests. This configuration is not suggested as a means of implementing tests which require the application of manually applied stimuli. For this would be using the PCM-Computer system strictly as a voltmeter and this is not considered to be a reasonable approach to implementation of such measurements. There are a total of 78 diagnostic measurements which do not require manually-generated stimuli and these 78 inputs are shown in the block diagram of the PCM-Computer configuration. To handle the other class of tests a VTVM is provided.

Figure 35 illustrates the other proposals. For instance, the Multimeter/Hardline/Test Panel would be represented by Figure 35 if the comparators were removed. For the Multimeter/Patch Cord/Test Panel, one must picture the comparators removed and a patch cord and connector in place of the large bundle of wires shown in the figure. The patch cord would consist of approximately 36 pairs of wires and would connect from the test location to the unit to be tested. It is estimated that this cord would be approximately 10 feet long and that three different types of patch cords would be required to cover the various systems. Further, it is assumed that there would be a minimum of two cords of each type.

3.2.2.2 Selection of Recommended System Confidence Tests

The guidelines used in selecting the recommended system for implementing confidence tests are shown in Figure 36. These guidelines are in addition to the obvious requirements of minimum weight, power, volume, maximum reliability, etc., which holds true for implementation of diagnostic tests as well as confidence tests. The multimeter approach does not satisfy the objectives of "providing quick fault detection" and "attracting the crew's attention". Therefore, this approach is considered unsatisfactory.

The PCM-Computer configuration can be ruled out on the "continuous availability" criterion, since there are periods during the mission when the computer will be fully occupied with functions other than checkout. Thus the only solution of the three proposed that will satisfy the confidence test guidelines is the comparator/display panel. To satisfy the guidelines for implementing confidence tests, the indications from these individual comparators would have to be mounted within easy view of the astronauts.

The recommended solution then for implementing the confidence tests is to use individual comparators for each test point and display the outputs from the comparators within easy view of the astronauts.

Diagnostic Tests

The four proposed solutions for implementing the diagnostic tests are again shown in Figure 37. The numbers shown immediately below the identity of the various systems represent estimates of the weights of the respective systems. As indicated in the figure, the weights range from a possible minimum of 15 pounds to a possible maximum of 35 pounds. The advantages and disadvantages listed in the figure are advantages or disadvantages of any one system with respect to the other three systems. A number of important factors, listed at the bottom of Figure 37, are deemed to have similar implications on all four systems. Particular note should be made of the fact that all four proposed systems include a multirange VTVM. It is also intended that the probes will exist to permit use of the VTVM on all GSE points which may be accessible, but are not included in the proposed implementation.

To select one of the four systems the following procedure was used:

1. Compare systems 4 and 3.

   Conclusion - The weight saving of system 4 over system 3 does not justify the cost of decreased reliability.

2. Compare systems 3 and 1.

   Conclusion - The advantage of system 3, namely simplicity, is not worth the weight differential. Further, one would probably have equally as complex a switching problem for system 3 as for system 1, since it would be desirable to be able to switch the VTVM across the comparator inputs as well as those inputs not fed to individual comparators.

3. Compare systems 1 and 2.

   Conclusion - System 1 is simple to use, provides faster fault isolation, and is more adaptable to sending the selected test data and its identity to the ground via the PCM system.

In view of the above conclusions, and under the assumptions that the PCM system will contain self-test circuitry as presently planned, system #1 is the recommended implementation of the diagnostic tests.

3.2.2.3 Further Considerations

Although the PCM-Computer configuration has not been recommended as the means of implementing either the confidence or the diagnostic tests, consideration should be given to the following arguments:

1. Many of the human engineering considerations have not been treated. When additional data become available it may be necessary or highly desirable to decrease the astronaut checkout tasks.

2. The role of the ground in assisting the crew in checkout activities has not been firmly established. It may be determined that the ground should not be slaved to the crew for information on diagnostic test measurements.

3. Overall system GO/NO-GO evaluations are made by the crew by mentally processing the information displayed on the flight panel. The PCM-Computer configuration can provide this facility easily, should further studies indicate a need for very quick overall evaluations at critical times in the mission. Similarly, detailed test information may be stored in the computer and displayed to the crew, either upon demand or if tolerances are exceeded.

4. The computer reliability may be increased to the point where it can justifiably be considered as a link in the checkout system.

Therefore, to leave the door open to the possible weight savings and increased capabilities of a PCM-Computer configuration, it is recommended that the provisions be made in PCM design to accept requests from the computer. The additional circuitry involved should not be very much, and it appears to be a small penalty to pay at this time for the ability of using the PCM-Computer configuration at a later time if deemed desirable.

4.0 Contrasts With NAA-IFTS

4.1 Integrated Test Equipment Concept

This report shows that integrated test equipment is not universally appropriate for either confidence or diagnostic tests. Only some of the CM equipments are suitably tested in this fashion. In particular, the PCM equipment and the guidance computer do not lend themselves to such tests, and the more basic proposition of whether or not these equipments should be maintained by testing and repair techniques has been challenged. If, indeed, the maintenance concept is tenable, then the present PCM testing approach, using self-test circuitry, is deemed the logical method of implementing tests for this equipment. No maintenance recommendations can even be made for the computer at this time.

4.2 Comparators and Display Lamps

Comparators and Display Lamps have been recommended in this report solely in connection with the implementation of confidence tests, as opposed to the NAA-IFTS concept for using comparators and display lights for both confidence and diagnostic tests.

4.3 Single Module Isolation

Rather than diagnosing malfunctions to within 2 or 3 modules, it has been the objective in this report to achieve single module isolation, excepting only those cases where it appears highly unreasonable to pursue this goal. This has increased the number of test points beyond those currently listed for IFTS display, particularly in the SCS.

4.4 Hardware For Integrated Equipment

This report recommends comparators and display lamps for confidence tests. For diagnostic tests a multi-range AC-DC voltmeter is judged most suitable. Requirements for signal conditioning and comparator mode logic are thereby reduced from that required in the NAA proposal which employs comparators and display lights for both sets of measurements.

4.5 Stimuli

Certain stimuli requirements have been recognized in this report. In the SCS, low-level (0 - 5 v) 400 cps ac and low-level dc voltages are appropriate. In the C&D subsystem modulated rf stimuli appear desirable.

5.0 Continuation of Study

Aside from certain general concepts, only the CM/SM system was treated in detail. In order to derive a complete set of in-flight checkout requirements for Apollo, the study will be extended to cover major items of interest which were omitted.

1. The Lunar Excursion Module

2. S-IVB/Instrument Unit

3. Radar and Digital Command Systems of the Command Module

A number of problem areas were also pointed out in the report as meriting more thorough analysis than was possible in the time available. Notably, aspects of human factors, sparing policies, subsystem criticality, to name a few, will be considered in the continuation of this effort.

#1

TYPICAL MISSION PROFILE

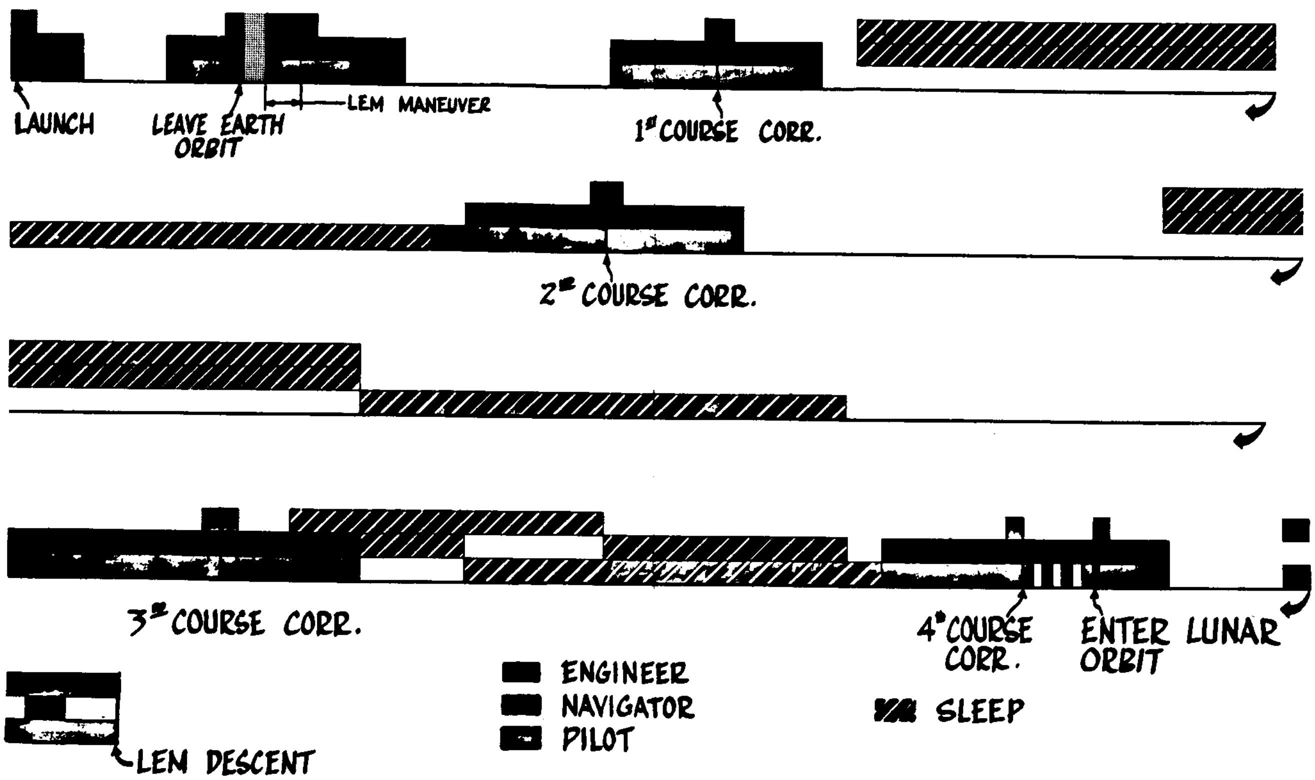

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Figure 1

#1

#2

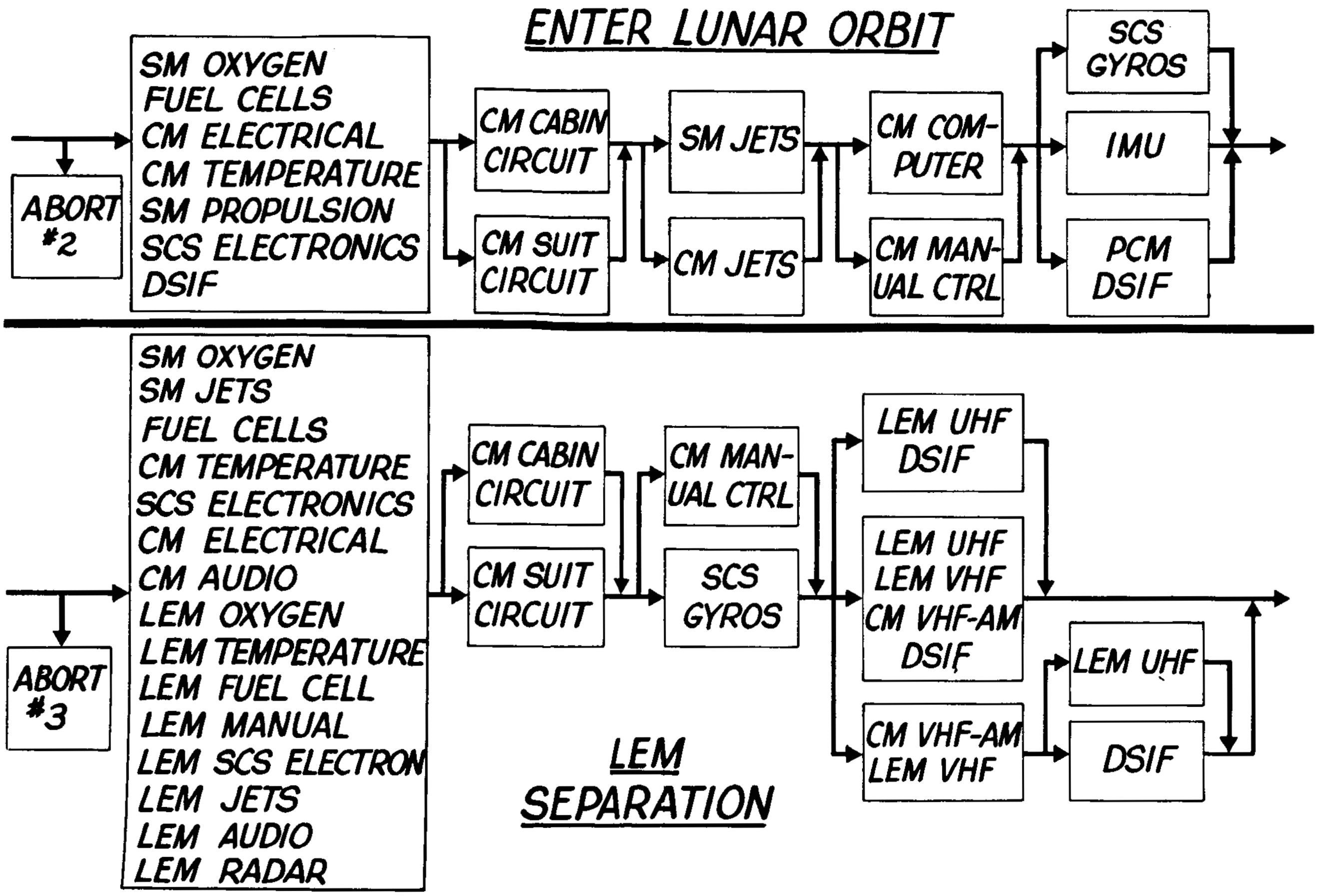

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Figure 2

#1

#2

TIMES AVAILABLE FOR CHECKOUT AND REPAIR PRIOR TO KEY MISSION EVENTS*

	AT COUCHES		AT BAY	
	MIN.	MAN-MIN.	MIN	MAN-MIN
1. LEAVE EARTH ORBIT	55	135	70	210
2. LEM MANEUVER	20	60	—	—
3. CORRECTION No. 1	165	260	175	525
4. CORRECTION No. 2	195	195	780	1170
5. CORRECTION No. 3	210	210	1620	3480
6. CORRECTION No. 4	210	210	540	540
7. ENTER LUNAR ORBIT	60	100	—	—
8. LEM DESCENT	40	40	180	270
9. LEAVE LUNAR ORBIT	40	40	120	360

*SIMILAR RETURN TIMES

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Figure 3

#1

#2.

SCS PITCH CHANNEL BLOCK DIAGRAM

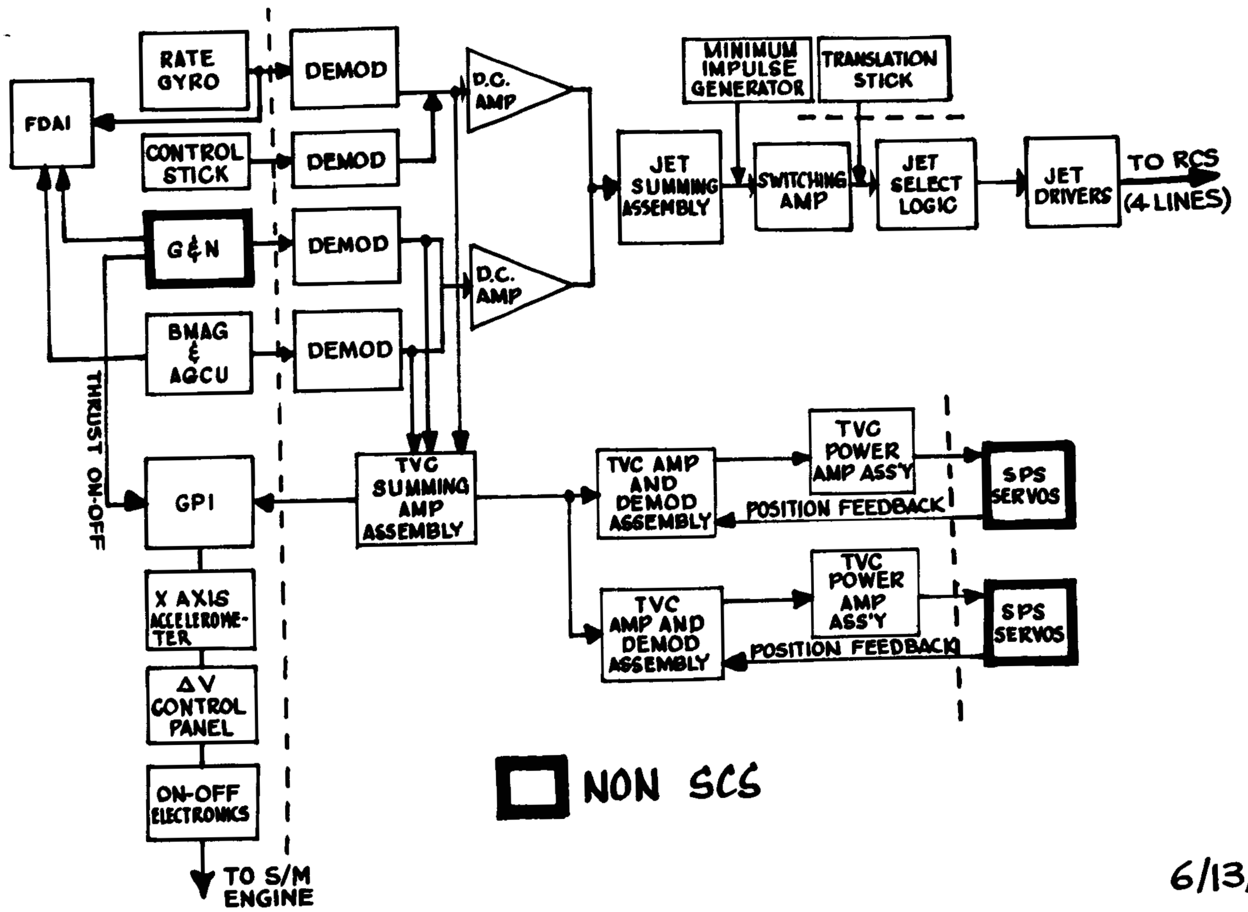

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Figure 4

#1

#2

SCS PHYSICAL CHARACTERISTICS

PACKAGE	APPROX WEIGHT	APPROX VOLUME (CU IN)	NO. OF MODULES	NO. OF TYPES OF MODULES	USAGE TIME %
AGAP	10	400	4	2	100
ECA-P	28	900	14	11	100
ECA-Y	28	900	14	11	100
ECA-R	28	875	10	8	100
ECA-D	30	1050	29	18	100
ECA-X	30	850	19	8	100
RGP	6	200	4	2	20
GPI	6.0	?	1	1	1.0
ΔV PANEL	5.0	?	1	1	1.0
CONTROL PANEL	7.0	300	1	1	100
MANUAL CONTROLS	19.0	?	–	–	–
FDAI	12.0	550	1	1	100
TOTAL	~ 200	?	~100	~ 50	

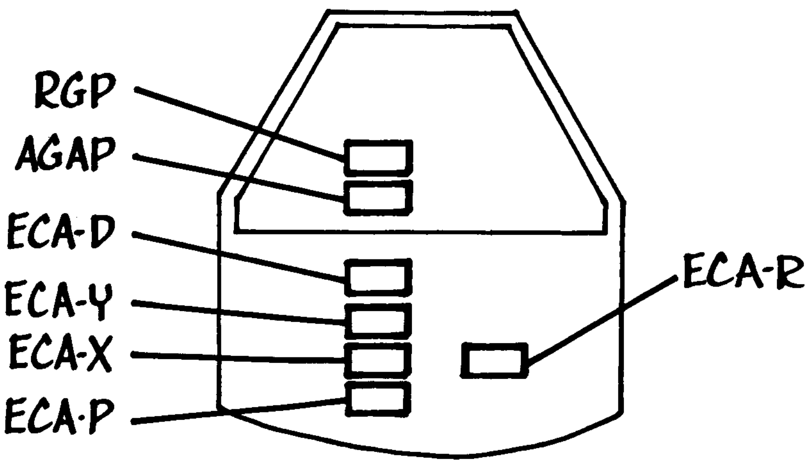

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Figure 5

#1

#2

CONFIDENCE TESTS

TEST	STATIC — STATUS SOURCE: FDAI	ΔV PANEL	GPI	NONE		DYNAMIC — STATUS SOURCE: FDAI	ΔV PANEL	GPI	NONE	
ATTITUDE CONT.										
PITCH, YAW ROLL RATE				X		X				
PITCH, YAW ROLL ATTITUDE				X		X				
BMAG REFERENCE				X		X				
G&N REFERENCE				X		X				
MINIMUM IMPULSE				X		X				
MAN. ATTITUDE CONT.				X		X				
X,Y,Z TRANS.				X			X*		X	*X AXIS ONLY
DEADBAND				X		X				
BMAG RATE BK-UP				X		X				
DISPLAYS										
FDAI				X?					X?	LIMITED REASON-ABLENESS TESTING IMPLICIT FROM OPERATION
ΔV INDICATOR				X?					X?	
GPI				X?					X?	
ΔV										
ACCELEROMETER & ELECTRONICS				X			X			
ENG. FIRE CIRCUITS				X			X			
G&N TVC			X					X		
SCS TVC			X					X		
RE-ENTRY										
SWITCHING & ROLL RATE SCALING FACTOR				X		X?				

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Figure 6

DIAGNOSTIC TESTS

PACKAGE	NO. OF TEST POINTS
ECA-P	31
ECA-R	36
ECA-Y	31
ECA-D	25
ECA-X	20
AGAP	4
RGP	4
TOTAL	151

ASSUMPTIONS:

- 1 MODULE ISOLATION UNLESS PARTICULARLY IMPRACTICAL
- LIMITED DEPENDENCE ON CORRELATION

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Figure 7

MEASUREMENT TYPES

	NO. OF POINTS
MEDIUM LEVEL DC-VARIABLE 0±6V	15
MEDIUM LEVEL AC-VARIABLE 0±6V	47
HIGH LEVEL DC-VARIABLE 0-28 VDC	44
HIGH LEVEL DC-CONSTANT >6V	11
MEDIUM LEVEL DC-CONSTANT 1 TO 6V	17
LOW LEVEL AC-CONSTANT <1V	9
PULSES	3

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Figure 8

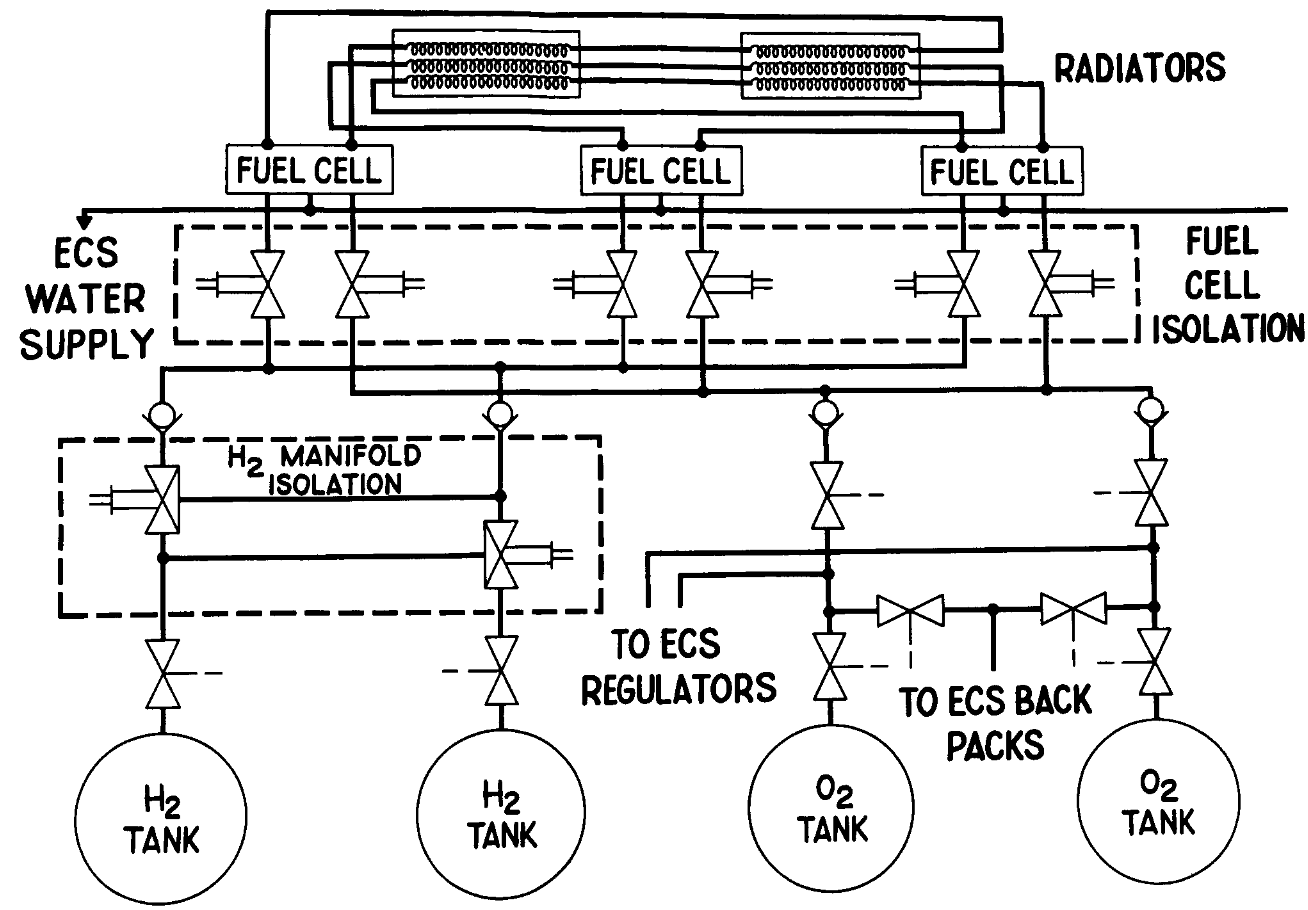

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Figure 9

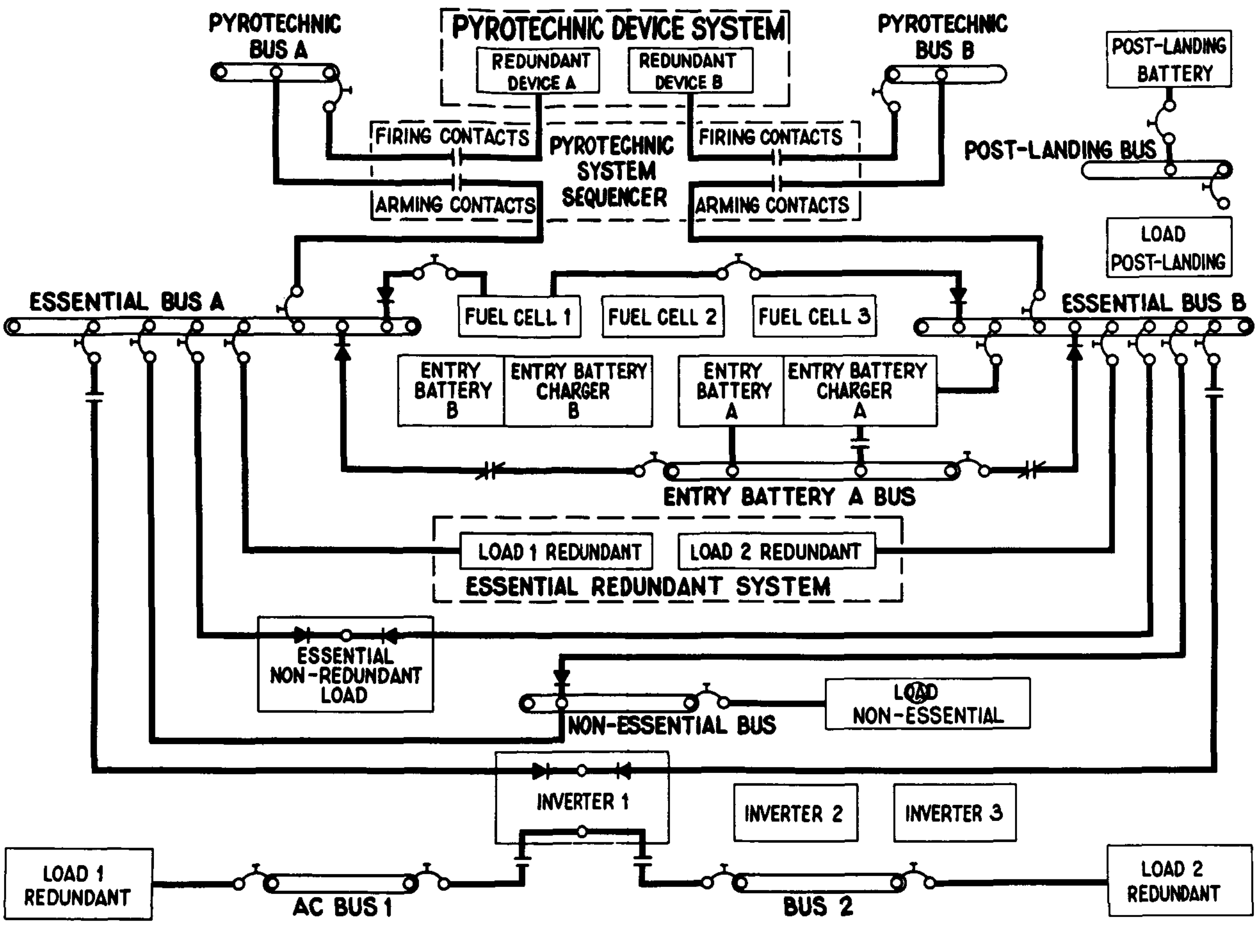

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Figure 10

TEST REQUIREMENTS

- FLIGHT INSTRUMENT PANEL
- CONDUCTIVE RF NOISE

SUMMARY

- EXTENSIVE REDUNDANCY (HEAVY UNITS)
  NO MAINTENANCE PLANNED
- OPPORTUNITIES FOR MAINTENANCE
- STUDY OF SPARES vs. REDUNDANCY
  e.g. INVERTERS.

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Figure 11

~~CONFIDENTIAL~~

COMM & DATA SUBSYSTEM NEAR-EARTH EQUIPMENT

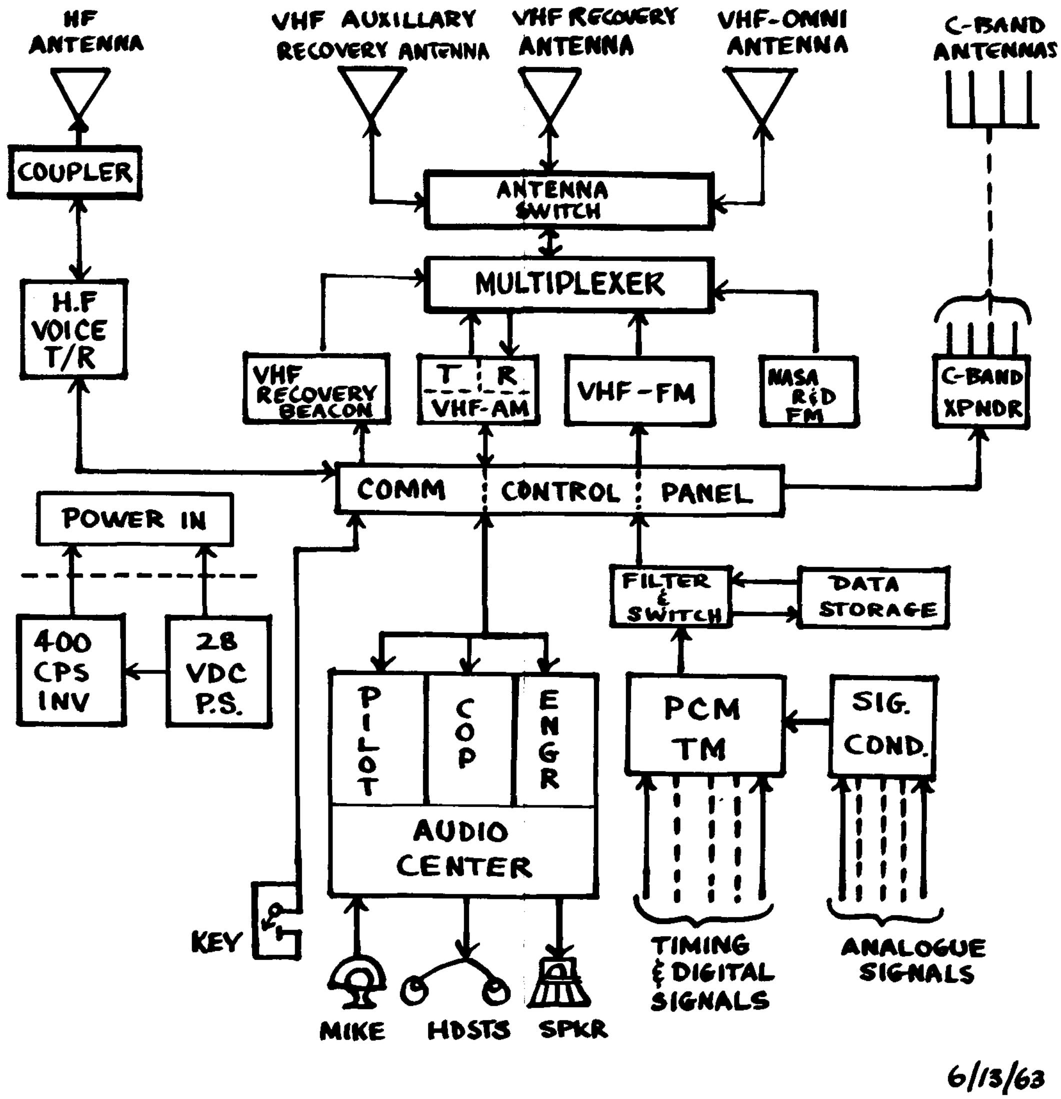

Figure 12

~~CONFIDENTIAL~~

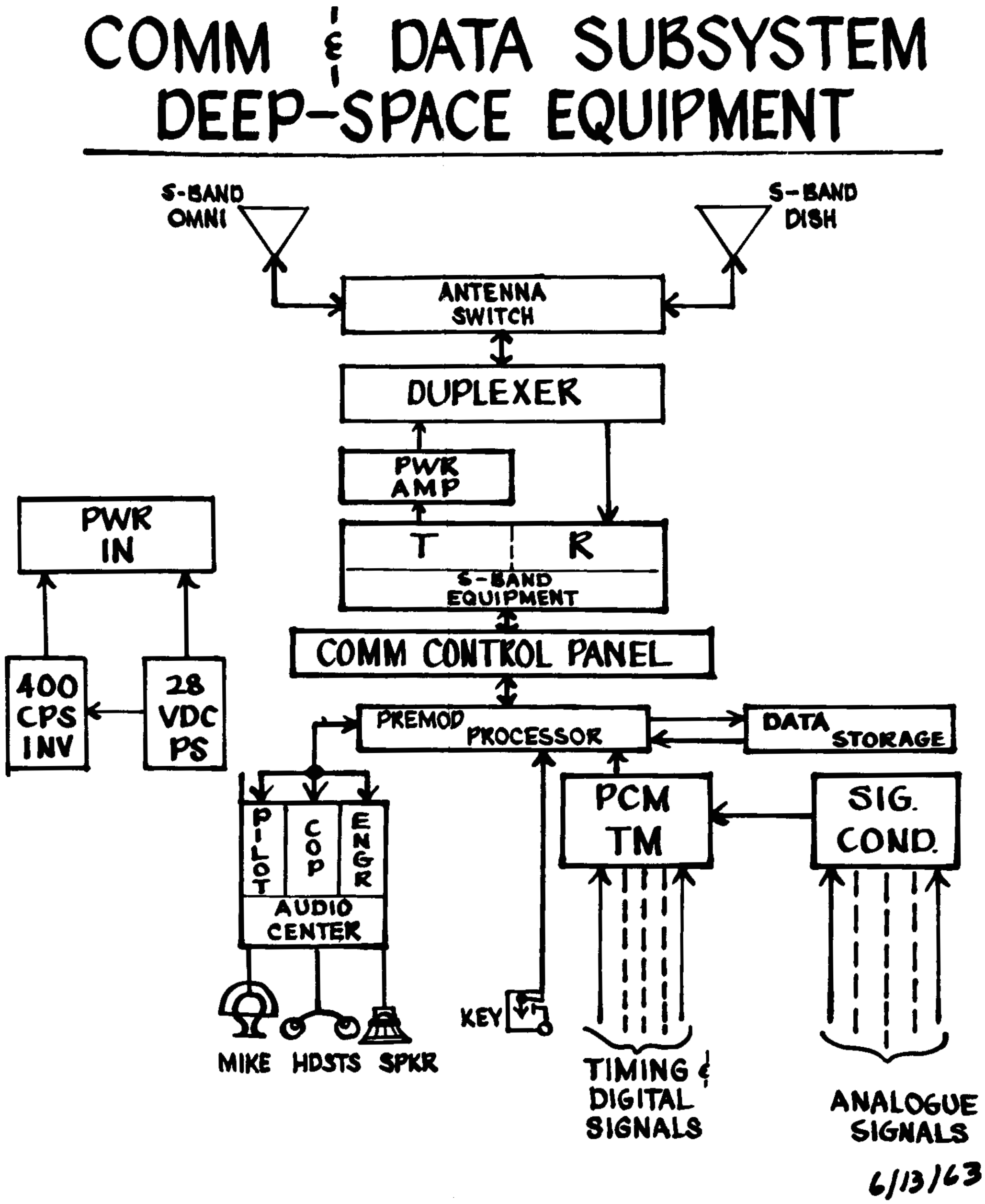
COMM & DATA SUBSYSTEM
DEEP-SPACE EQUIPMENT
S-BAND OMNI
S-BAND DISH
ANTENNA SWITCH
DUPLEXER
PWR AMP
PWR IN
T
R
S-BAND EQUIPMENT
COMM CONTROL PANEL
400 CPS INV
28 VDC PS
PREMOD PROCESSOR
DATA STORAGE
PILOT
COP
ENGR
AUDIO CENTER
PCM TM
SIG. COND.
MIKE
HDSTS
SPKR
KEY
TIMING & DIGITAL SIGNALS
ANALOGUE SIGNALS
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Figure 13

UNDEFINED EQUIPMENT

- CM-LEM COMM. EQUIPMENT
  - ✓ VOICE
  - ✓ RADAR
  - ✓ ETC.
- UP-DATA (DIGITAL COMMAND SYSTEM)
- ANTENNA SYSTEMS
  - ✓ VHF OMNI
  - ✓ VHF RECOVERY
  - ✓ VHF AUX. RECOVERY
  - ✓ HF ANTENNA
  - ✓ S-BAND OMNI
  - ✓ S-BAND DIRECTIONAL
- NASA R & D FM XMTR

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Figure 14

PHYSICAL CHARACTERISTICS

SUBSYSTEM EQUIPMENT	WEIGHT (LBS)	VOLUME (in^3)	# OF MODULES	#MODULE TYPES	APPROX. USAGE TIME	PREDICTED RELIABILITY	ASSIGNED RELIABILITY
S-BAND TRNSPNDR.	17	1017	6	3	98%	0.9898	0.9990
S-BAND PWR. AMP.	18	570	1	1	98%	0.9932	0.9992
PCM-TM	40	1618	42	22	100 %	0.9675	0.9970
SIGNAL CONDITIONERS	31	675	44	7	100 %	0.9741	0.9996
AUDIO CENTER	22	1070	3	1	100 %	0.9959	0.9995
PREMOD PROC.	10	377	6	6	100 %	0.9948	0.9995
DATA RECORDER	22	938	1	1	UNK	0.9925	0.9930
TV CAMERA	UNK	UNK	UNK	UNK	UNK	UNK	UNK
C-BAND TRNSPNDR.	20	570	1	1	2 %	0.9991	0.9999
VHF-AM EQUIP.	11	420	3	3	17 %	0.9991	0.9999
VHF-FM XMTR.	8	265	2	2	2%	0.9998	0.9999
VHF-RCVY BCN.	6	140	1	1	15%	0.9984	0.9999
VHF MUX	15	656	1	1	17%	UNK	UNK
HF TRNSCVR.	11	265	4	4	15%	0.9995	0.9999
TOTALS	231+	≃8ft.^{3}	115+	53+			

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Figure 15

COMMUNICATIONS & DATA SUBSYSTEM-LOWER EQUIP. BAY

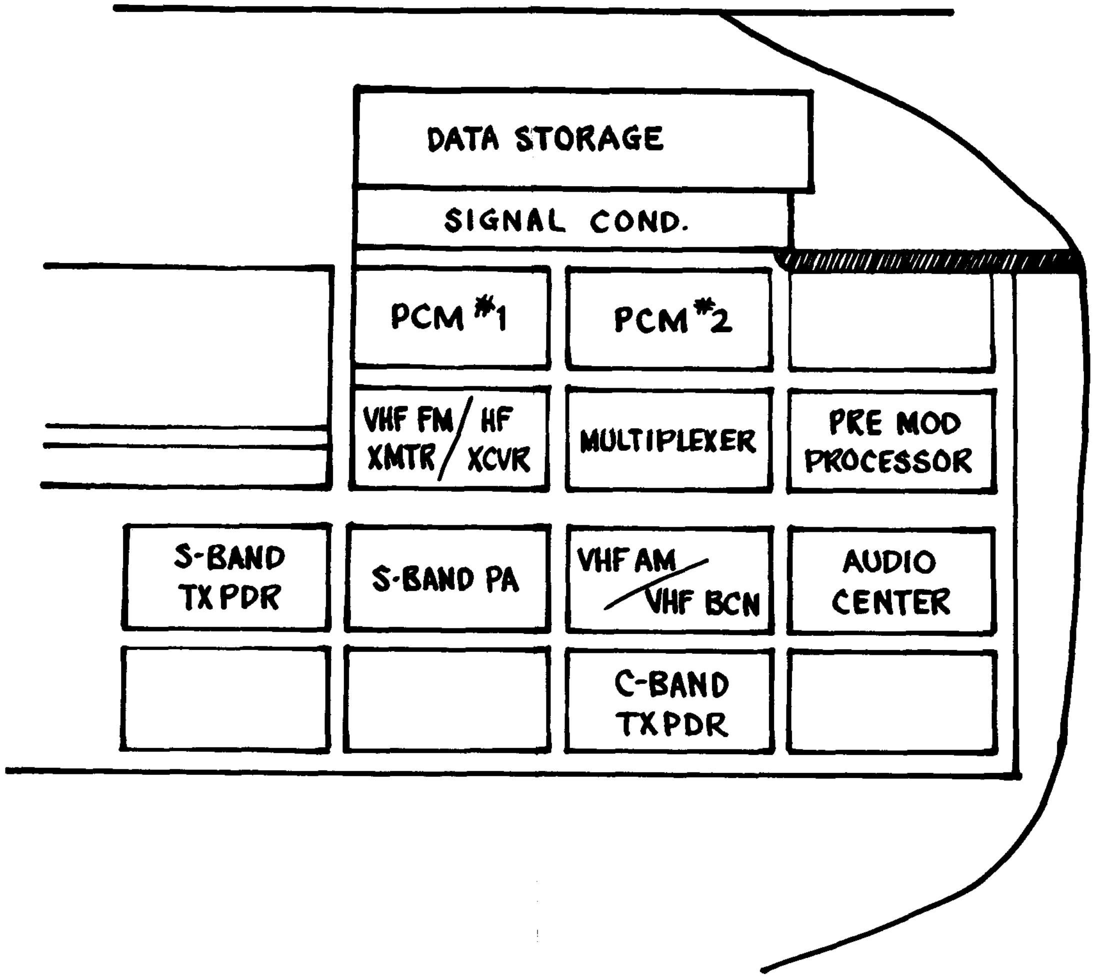

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Figure 16

ALTERNATE MODES	1st ALTERNATE CAPABILITY	OTHER ALTERNATE CAPABILITIES
S-BAND TRANSPONDER	REDUNDANT BACK-UP SYS.	(1) REPAIR WITH SPARES (2) HF TRANSC. VOICE BACK-UP
S-BAND PWR. AMPLIFIER	REPAIR	BYPASS
PCM-TM/SIG. COND.	REPAIR / DELETE CHANNEL-REPATCH SIGNAL IF DESIRED	(1) AUDIO REPORTING OF S/C INFORMATION / NONE
AUDIO CENTER	REDUNDANT BACK-UP SYS.	(1) REDUNDANT BACK-UP SYST. (2) REDUNDANT BACK-UP SYST. (3) KEY
PREMOD PROCESSOR	REPAIR	BYPASS WHERE POSSIBLE
DATA STORAGE RECORDER	REPAIR	NONE
TV CAMERA	AUDIO REPORTING OF VISUAL DATA	NONE
CM-LEM COMM.	UNDEFINED	UNDEFINED
C-BAND TRANSPONDER	S-BAND EQUIPMENT	REDUNDANT S-BAND EQUIPMENT
VHF-AM EQUIPMENT	REPAIR	(1) HF TRANSCEIVER (2) VHF-FM TRANSMITTER (3) S-BAND EQUIPMENT
VHF-FM TRANSMITTER	REPAIR	(1) AUDIO REPORTING OF SPACECRAFT [illegible] (2) S-BAND EQUIPMENT (3) VHF-AM ON LOW-DATA RATE
VHF RECOVERY BEACON	KEYED VHF-AM	KEYED VHF-FM KEYED S-BAND
OP. MULTIPLEXER	BYPASS, IF POSSIBLE	NONE
HF TRANSCEIVER	VHF-AM EQUIPMENT	(1) VHF-FM TRANSMITTER (2) S-BAND EQUIPMENT

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CONFIDENCE CHECKOUT BY MISSION PHASE

	1 PRIOR TO LUNAR TRANSFER INJECTION	2 PRIOR TO DOCKING MANEUVER	3 PRIOR TO MIDCOURSE MANEUVERS	4 PRIOR TO PERILUNE DEBOOST	5 PRIOR TO LEM SEPARATION	6 AFTER LEM SEPARATION	7 AFTER LUNAR LANDING	8 PRIOR TO LUNAR ASCENT	9 PRIOR TO EARTH TRANSFER INJECTION	10 PRIOR TO RE-ENTRY
S-BAND TRANSPONDER	XMTR-PWR RCVR-AGC VOICE-PRESENCE	→	→	→	→	--	XMTR-PWR RCVR-AGC VOICE-PRESENCE	→	→	
S-BAND PWR AMPLIFIER	OUTPUT PWR ALL MODES	→	→	→	→	--	OUTPUT PWR ALL MODES	→	--	
PCM-TM/SIG COND.	SELF CHECK VERIFICATION FROM GROUND	→	→	→	→	--	SELF CHECK VERIFICATION FROM GROUND	→	→	
AUDIO CENTER	VOICE CHECK TO GROUND ALL CHANNELS	→	→	→	→	--	VOICE CHECK TO GROUND ALL CHANNELS	→	→	
PREMOD PROCESSOR	CHECK ONLY WHEN AN S-BAND FCN FAILS	→	→	→	→	--	→	→	→	
DATA STORAGE RECORDER	TEST BY USAGE	--	--	TEST BY USAGE	→	--	--	--	--	
TV CAMERA	TEST BY USAGE	--	--	TEST BY USAGE	→	--	--	--	--	
CM-LEM COMM.	--	--	--	TEST BY USAGE	→	→	→	--	--	
C-BAND TRANSPONDER	TEST BY USAGE	--	--	--	--	--	--	--	--	XMTR-PWR RCVR-OUTPUT
VHF-AM EQUIPMENT	TEST BY USAGE	--	--	--	--	--	--	--	--	XMTR-PWR RCVR-AUDIO
VHF-FM TRANSMITTER	TEST BY USAGE	--	--	--	--	--	--	--	--	XMTR-PWR
VHF RECOVERY BEACON	TEST BY USAGE	--	--	--	--	--	--	--	--	TEST BY USAGE (GRND. CONFIRM)
OPERATIONAL MULTIPLEXER	TESTED IMPICITLY	--	--	--	--	--	--	--	--	TESTED IMPLICITLY
HF TRANSCEIVER	TEST BY USAGE	--	--	--	--	--	--	--	--	XMTR-PWR RCVR-AUDIO

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Figure 18

EQUIPMENT LEVEL TESTING

EQUIPMENT	SUFFICIENT CONFIDENCE INFORMATION	ASSUMED INFO
XMTRS	Output Waveform Characteristics • Carrier Amplitude • Carrier Frequency • Modulation Waveform	Carrier Amplitude
RCVRS	Demodulation Waveform • Fidelity • S/N Ratio	Positive Decoding Action (AGC)
SIGNAL PROCESSORS	Transfer Function • Frequency Response • Phase Characteristic	Attenuation
PCM	Individual Channel Responses	SELF-TEST CAPABILITY

NOTE: All confidence tests require stimuli.

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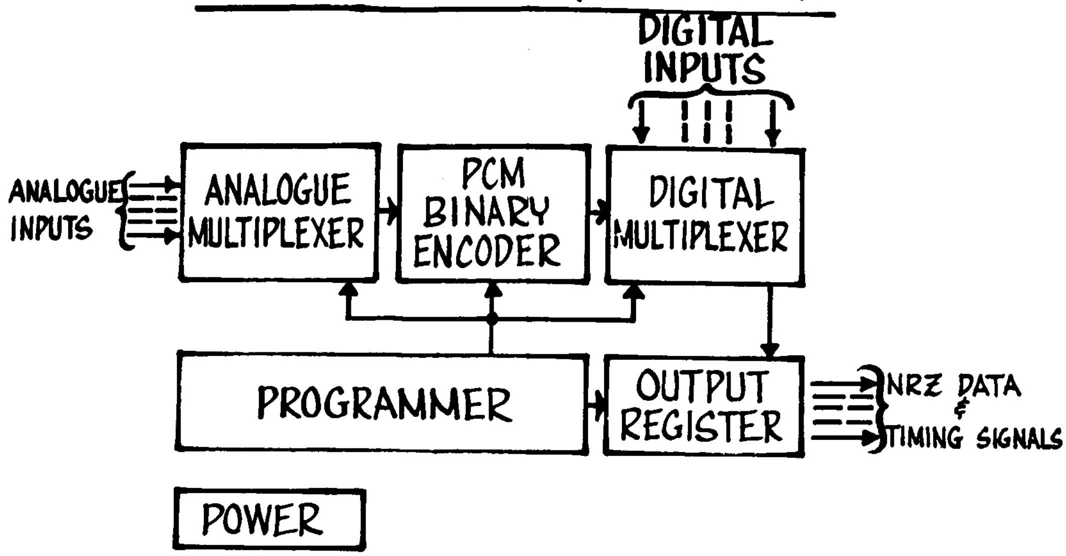
Figure 19

BASIC PCM EQUIPMENT



FUNCTIONAL BLOCK	# OF MODULES	# MODULES TESTED
ANALOGUE MULTIPLEXERS	18	3
PCM BINARY ENCODER	2	2
DIGITAL MULTIPLEXER	6	6
OUTPUT REGISTER	1	1
PROGRAMMER	11	5
POWER SUPPLY	3	3
TOTALS	41	20

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Figure 20

TEST POINTS REQ'D

EQUIPMENT	NO. OF TEST POINTS			
	WITH GRND.		W/O GRND.	
	CONF.	DIAG.	CONF.	DIAG.
S-BAND XPNDR	0	2	2	0
S-BAND P.A.	1	0	1	0
PCM	0	0	>50	0
SIG. COND.	0	2	>250	0
AUD. CNTR.	0	0	0	0
PMP	0	0	15	4
DSE	0	0	0	0
TV	0	0	1	0
C-BAND XPNDR	0	0	2	0
VHF-AM	0	2	2	2
VHF-FM	0	4	1	4
VHF-BCN	0	0	1	0
MUX	0	0	6	0
HF	0	3	2	3
TOTALS	1	13	12+	13

ASSUMPTIONS:

- ANTENNAS WORKING, GROUND ALWAYS TRANSMITTING
- TROUBLE SHOOTING AID
- SINGLE MODULE ISOLATION, IN MOST CASES

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Figure 21

TEST POINTS REQ'D

EQUIPMENT	NO. OF TEST POINTS: WITH GRND.		W/O GRND.	
	CONF.	DIAG.	CONF.	DIAG.
S-BAND XPNDR			2	1
S-BAND P.A.			1	0
PCM			N/A	N/A
SIG. COND.			N/A	N/A
AUD. CNTR.			0	0
PMP			15	4
DSE			0	0
TV			0	0
C-BAND XPNDR			2	0
VHF-AM			2	3
VHF-FM			1	4
VHF-BCN			1	0
MUX			6	0
HF			2	4
TOTALS			32	16

ASSUMPTIONS:

- ANTENNAS WORKING, GROUND ALWAYS TRANSMITTING
- TROUBLE SHOOTING AID
- SINGLE MODULE ISOLATION, IN MOST CASES

Figure 22

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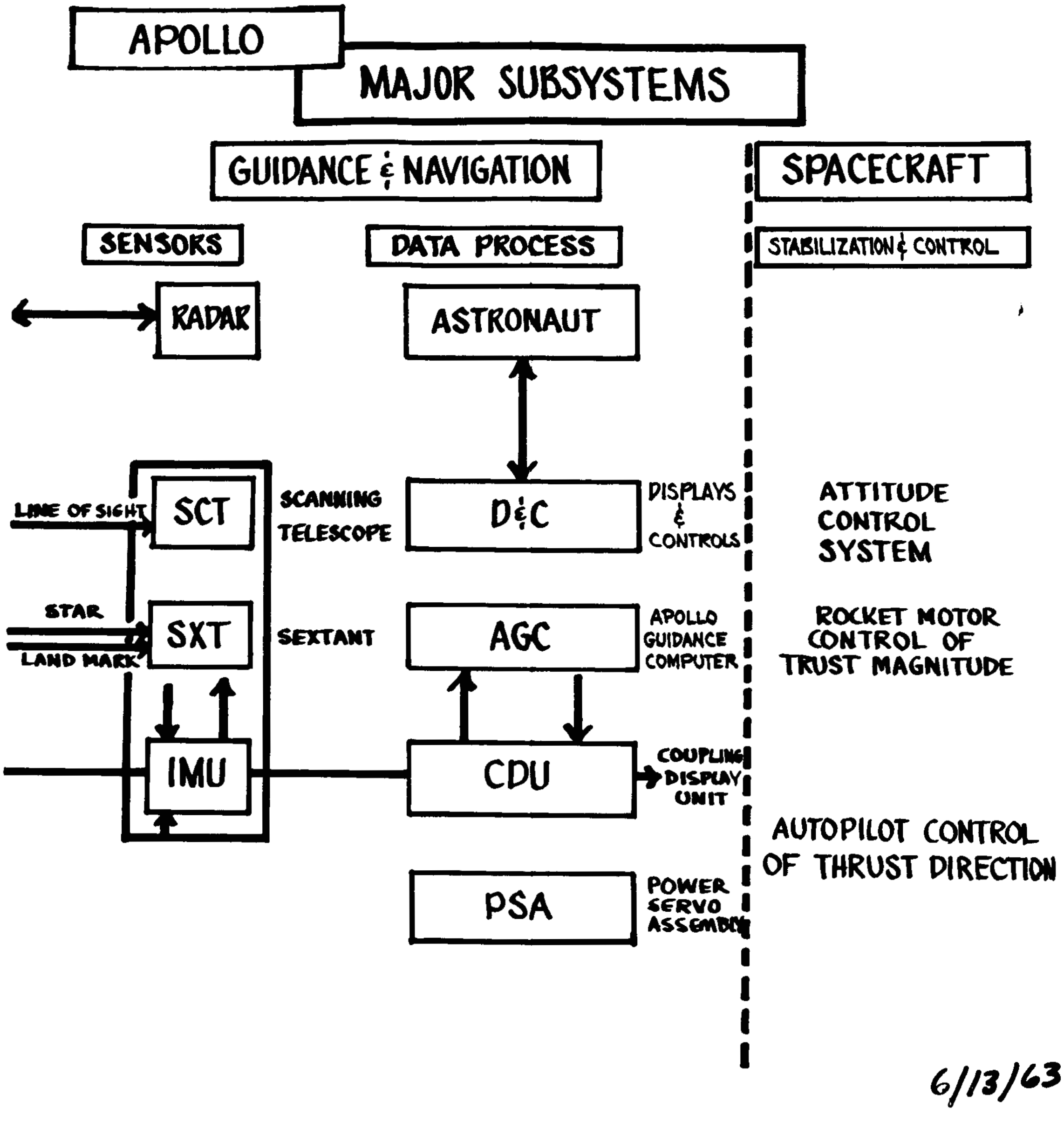
APOLLO
MAJOR SUBSYSTEMS
GUIDANCE & NAVIGATION
SPACECRAFT
SENSORS
DATA PROCESS
STABILIZATION & CONTROL
RADAR
ASTRONAUT
LINE OF SIGHT
SCT
SCANNING TELESCOPE
D&C
DISPLAYS & CONTROLS
ATTITUDE CONTROL SYSTEM
STAR
LAND MARK
SXT
SEXTANT
AGC
APOLLO GUIDANCE COMPUTER
ROCKET MOTOR CONTROL OF TRUST MAGNITUDE
IMU
CDU
COUPLING DISPLAY UNIT
AUTOPILOT CONTROL OF THRUST DIRECTION
PSA
POWER SERVO ASSEMBLY
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Figure 23

THE G&N SYSTEM

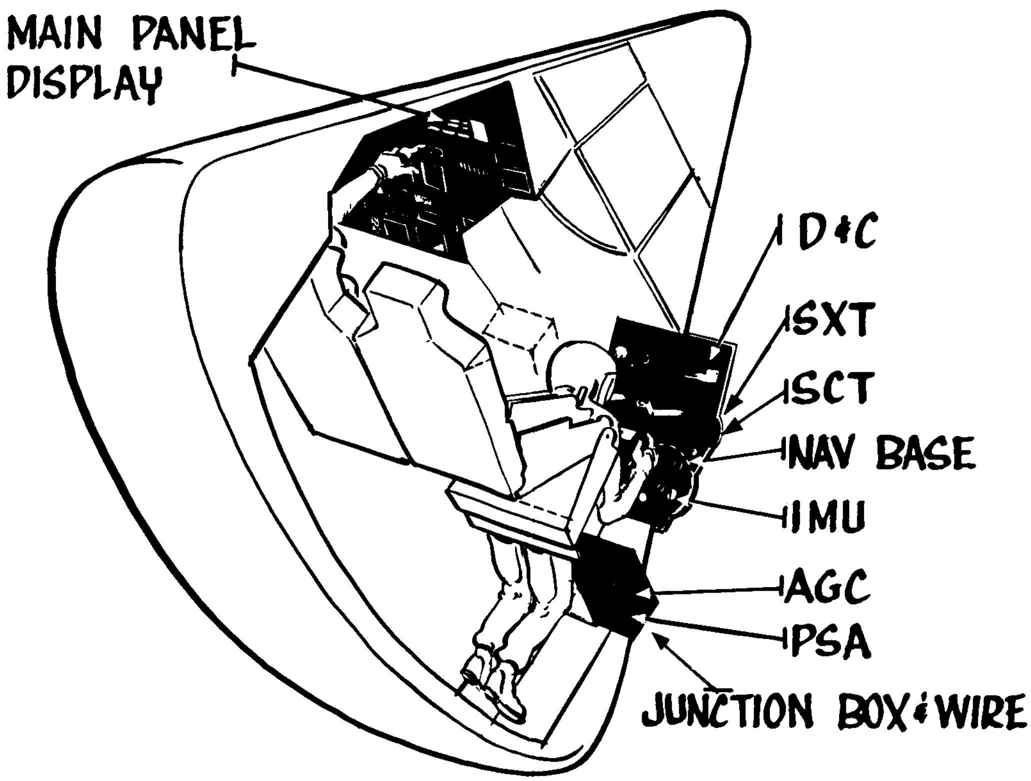

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Figure 24

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PHYSICAL CHARACTERISTICS

	WEIGHT	POWER	MTBF	DUTY CYCLE	NO. OF MODULES REPAIRABLE
IMU	67 LBS.	273 WTS.	5900 HRS.	15 %	NO —
OPTICS	32	134	10,300	25%	NO —
PSA	50	N/A	2600	N/A	YES 90
AGC	93	120	940	33%	YES 75
DISPLAY	78	≈15	?	N/A	NO
NAV. BASE & WIRING	85	N/A	?	N/A	NO
	475 LBS.				

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Figure 25

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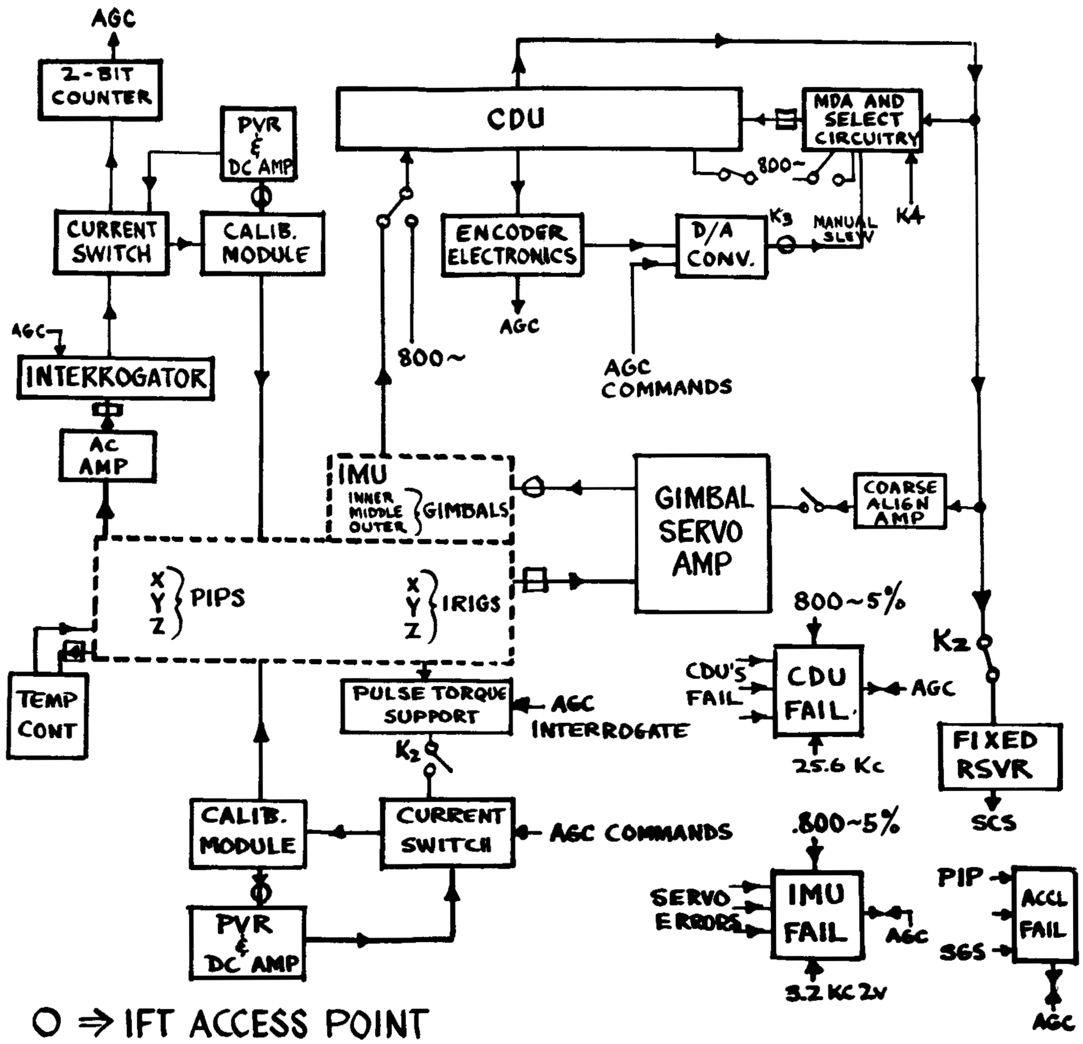
BLOCK I PACKAGING OF INERTIAL PSA



O ⇒ IFT ACCESS POINT

□ ⇒ IFTS MONITOR POINT

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Figure 26

APOLLO LOGIC DESIGN

CONTROL

- SINGLE-ADDRESS
- 2 MODES

		OPERATION	ADDRESS	
"NORMAL"		3	12	8 INSTRUCTIONS
"EXTENDED"	STEP 1:	OP1	ADDR. 1	3 INSTRUCTIONS
	STEP 2:		ADDR. 2	(MPY, DIV, SUB)

+

RESULT: OVER FLOW | NEW ADDRESS

NEW OPERATION

- INTERRUPT
- 4 I-ϕ INSTRUCTIONS

ARITHMETIC

- PARALLEL — SINGLE PRECISION: SIGN [1] MAGNITUDE [14] — 1 PART IN 16,000
- FIXED POINT — DOUBLE PRECISION: SIGN [1] MAGNITUDE [28] — 1 PART IN 256 MILLION
- TYPICAL TIMES (μ SEC)

		S.P.	D.P.	INTERPRETIVE
X = A+B	ADD	94	188	3360
X = A·B	MPY	210	996	4320
X = A/B	DIV	304	1147	5630

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Figure 27

APOLLO MEMORY

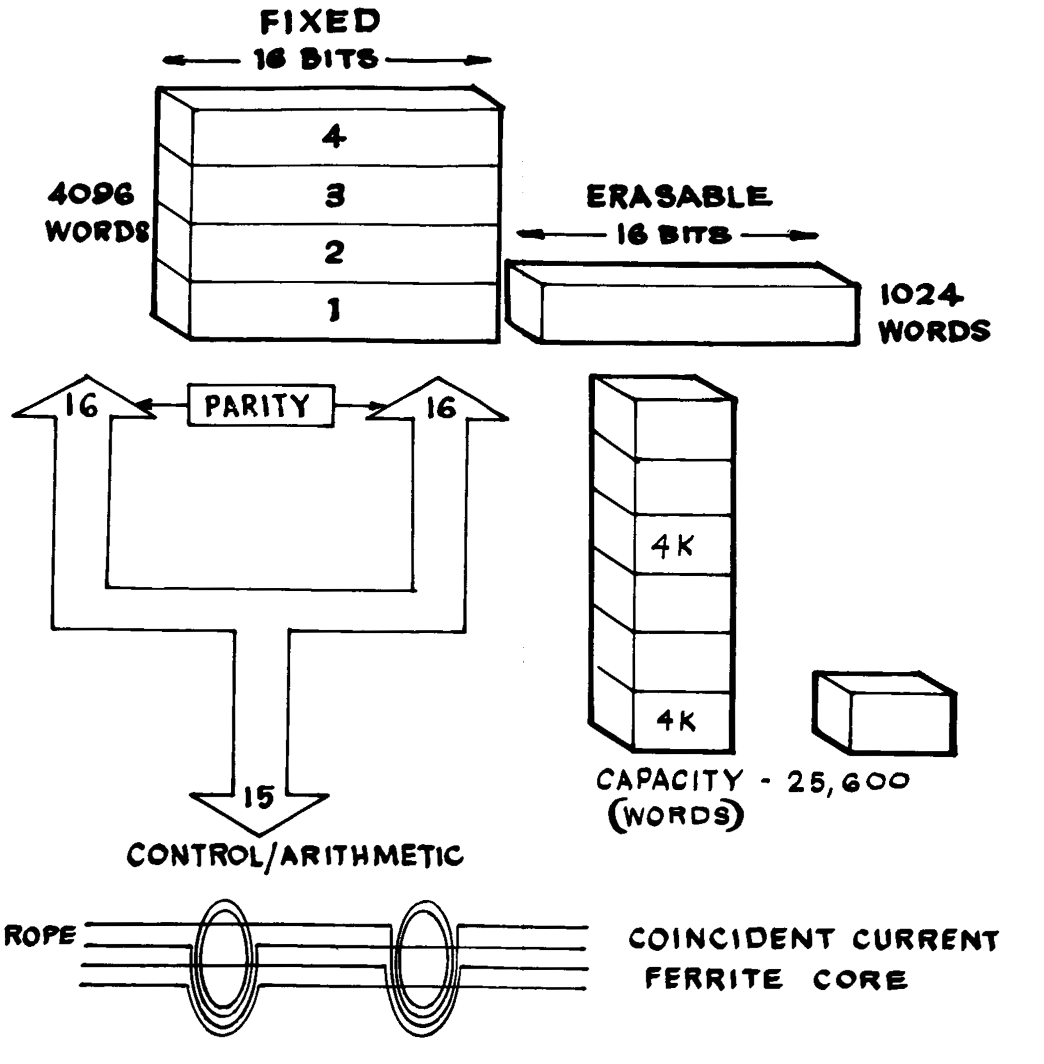

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Figure 28

AGC SELF-CHECKING FEATURES

- PARITY
- POWER FAIL
- TC TRAP; RUPTLOCK, COUNTERLOCK
- INACTIVITY, SCALER FAIL, 1.5 pps

- TEST PROGRAM
- INPUT-OUTPUT ECHO CHECKS

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Figure 29

CHARACTERISTICS

SYSTEM	WEIGHT		VOL	CHANNELS		POWER	BIT RATE	RELIABILITY
	ELECT.	MECH		ANAL.	DIGITAL			(MTBF)
TELSTAR (NON MODULAR)	7	1	242 IN.^{3}	104	16	<400 MW	16 B/S	70,400
SPACECRAFT (MODULAR)	14	36	1618 IN.^{3}	270	50	8 WATTS	51.2 KB/S	10,300

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Figure 30

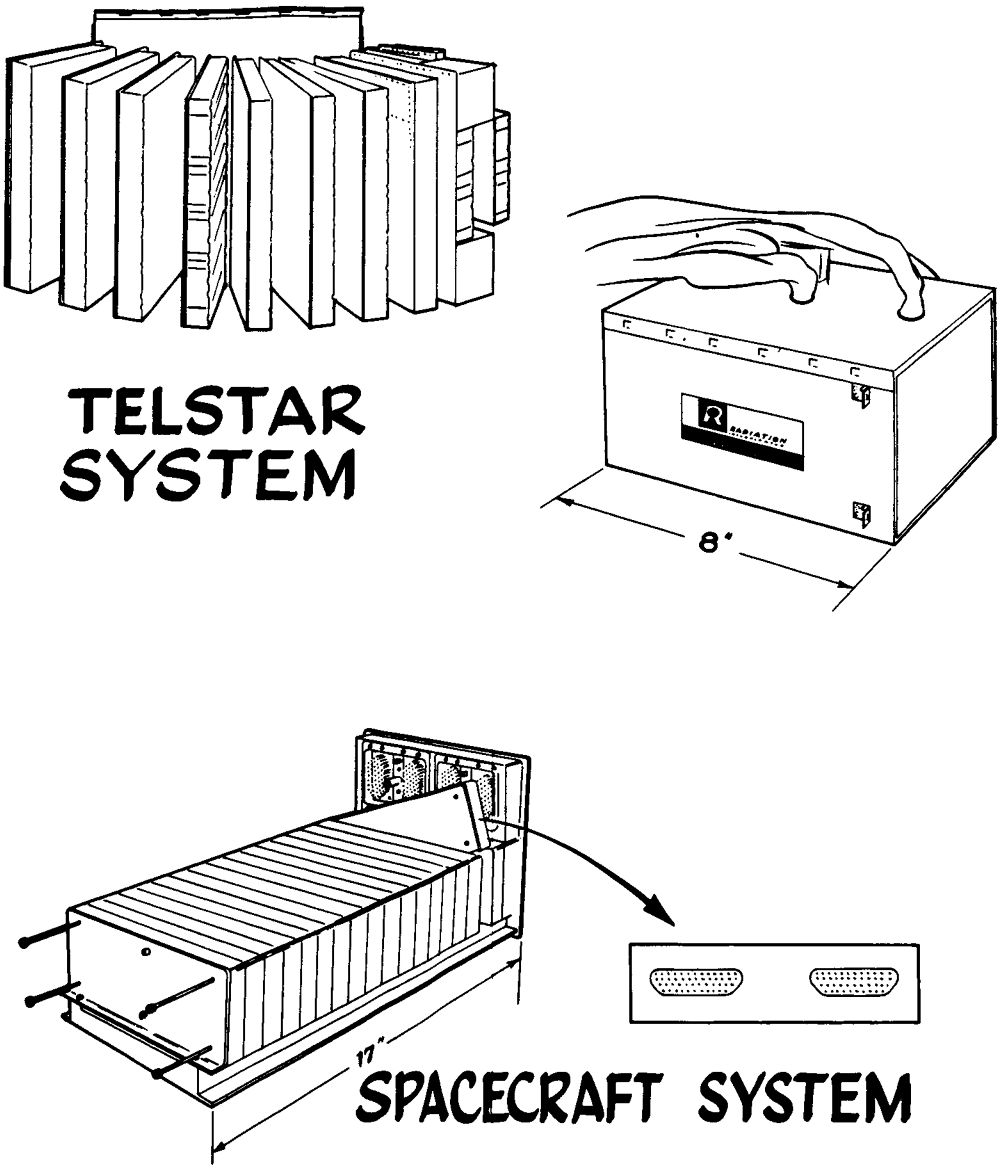

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Figure 31

TEST POINTS

GOAL:

1. PROVIDE INSTANTANEOUS STATUS INDICATIONS.
2. ISOLATE TO SINGLE MODULE IN MOST CASES

IN ADDITION TO FLIGHT DISPLAYS

SYSTEM	CONFIDENCE POINTS	DIAGNOSTIC POINTS
SCS	18	125
G&N		
PSA	0	57
AGC	0	4
COMM.	16	26
TOTAL	34	212

SPECTRUM OF MEASUREMENTS:

AC
DC
FIXED LEVEL
VARIABLE LEVEL
BI-LEVEL

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Figure 32

ALTERNATIVE TEST SYSTEMS

CONFIDENCE TESTS (34 TEST POINTS)	MULTIMETER COMPARATORS/DISPLAY PANEL PCM -COMPUTER
DIAGNOSTIC TESTS (212 TEST POINTS)	MULTIMETER/HARDLINE/TEST PANEL MULTIMETER/COMPARATORS/HARDLINE/TEST PANEL MULTIMETER/PATCH CORD/ TEST PANEL PCM-COMPUTER/VTVM

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Figure 33

PCM-COMPUTER CONFIGURATION

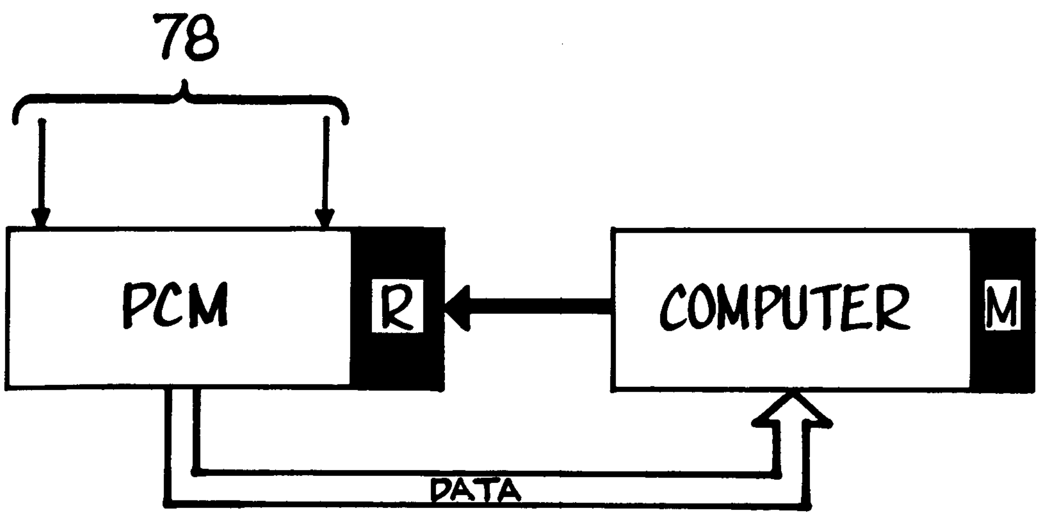

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Figure 34

PICTORIAL OF IMPLEMENTATION SCHEMES

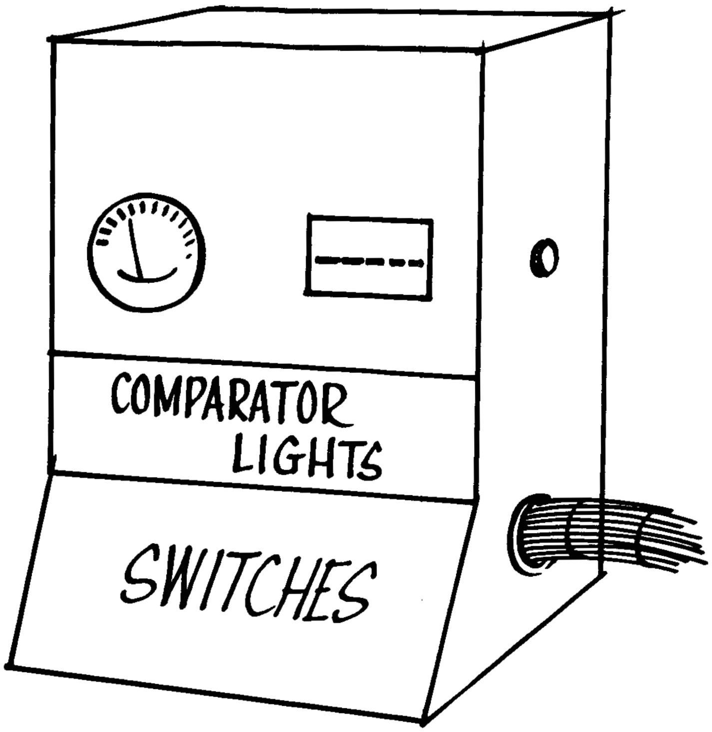

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Figure 35

CONFIDENTIAL TEST GUIDELINES

- QUICK FAULT DETECTION
- FEASIBLE WHILE CREW RESTRAINED
- ATTRACT CREW ATTENTION
- CONTINUOUS AVAILABILITY

RECOMMENDED SYSTEM

COMPARATORS/DISPLAY PANEL

- WEIGHT ESTIMATE : 5-10 lbs.

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Figure 36

ALTERNATIVE DIAGNOSTIC TEST SOLUTIONS

1 HARDLINE/VTVM PANEL 15-25	2 PATCH CORD VTVM/PANEL 16-25	3 COMPARATOR/HARDLINE PANEL/VTVM 30-45	4 PCM COMPUTER HARDLINE/PANEL/VTVM 25-35
	ADVANTAGES MAXIMUM GROWTH POTENTIAL	**ADVANTAGES** SIMPLEST TO USE	**ADVANTAGES** AUTOMATIC DATA TRANSMISSION TO GROUND
DISADVANTAGES COMPLEX SWITCHING	**DISADVANTAGES** MORE MOVEMENT REQUIRED RESTRICTED SEQUENCING	**DISADVANTAGES** HEAVIEST	**DISADVANTAGES** LEAST RELIABLE
ADVANTAGES OVER 2: SIMPLER TO USE FASTER FAULT ISOLATION MORE ADAPTABLE TO PCM		ADVANTAGE RELATIVE TO 1 NOT WORTH WEIGHT COST	WEIGHT SAVING RELATIVE TO 3 NOT WORTH RELIABILITY COST

TEST SYSTEM SIMILARITIES

SKILL REQUIRED OF CREW
MAINTENANCE TIME
AVAILABILITY OF PACE TEST POINT USING PROBE

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Figure 37